UNITED STATES OF AMERICA
CIVIL AERONAUTICS BOARD
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AIRWORTHINESS REQUIREMENTS FOR AIRCRAFT

CHAPTER I. GENERAL

Section 1. Scope of Requirements

(*A*) The requirements contained herein are those dealing with the design, construction, and flight characteristics of aircraft for use in air commerce. They are supplemental to and a part of the Air Commerce Regulations specified in Aeronautics Bulletin No. 7 and are supplemented by Aeronautics Bulletin No. 26, Design Information for Aircraft, which contains explanatory matter and useful information. Other airworthiness requirements are contained in the following publications:

Aeronautics Bulletin No. 7.-F— Airworthiness Requirements for Aircraft Components and Accessories
Aeronautics Bulletin No. 7-G — Airworthiness Requirements for Engines and Propellers
Aeronautics Bulletin No. 7-H — Air Commerce Regulations Governing Alterations and Repairs to Licensed Aircraft
Aeronautics Bulletin No. 7-J — Special Requirements for Transport Aircraft.

(*B*) These requirements are based on the present development in the science of airplane design. Experience indicates that, when applied to conventional types of construction, they will result in an airworthy and well-proportioned aircraft. New types of aircraft and new types of construction may, however, incorporate features to which these requirements cannot be logically applied. In such cases, special consideration will have to be given the particular new problems involved. In cases where the deviation from conventional practice is small, approval may be granted if sufficient evidence is submitted to show that the proposed deviation will not be detrimental to the airworthiness of the design. When the deviation from conventional practice is considerable, a special aircraft license may be granted pending a thorough study of the principles involved. Such aircraft will, by their very nature, be experimental and cannot be licensed for general commercial use until their airworthiness has been established.

(*C*) *Lighter-than-air aircraft.* — These requirements do not include regulations pertaining to lighter-than-air aircraft. Until sufficient data are available from which to formulate general regulations in this field, each design of a lighter-than-air aircraft will be handled by special rulings.

(*D*) *Gliders.*— Because of the wide variation in type and purpose of modern gliders, it is considered inadvisable to include herein a specific set of regulations covering this class of aircraft. When approval of a glider is desired preliminary data should be submitted outlining the type, method of construction, approximate dimensions and characteristics, method of launching, maximum towing speed, and other pertinent information. Upon receipt of such data, the necessary requirements will be determined.

Sec. 2. Aircraft Classification

Aircraft are divided herein into three classifications as follows:

(A) Normal airplanes, which are all heavier-than-air aircraft not included in definitions (B) and (C) below.

(B) Light airplanes, which are heavier-than-air aircraft of less than 1,000 pounds gross weight with a wing loading of not more than 6 pounds per square foot.

(C) Gliders, which are heavier-than-air aircraft having no power unit. (See sec. 1 (D).)

Sec. 3. Airworthiness Factors

In determining the airworthiness of aircraft the following factors are taken into consideration:

(A) The structural strength of wings, control surfaces; fuselage, engine mount and/or nacelles, fittings, control system, and landing gear.

(B) Cockpit, cabin, and control arrangements.

(C) Power plant and power plant installation.

(D) Equipment and instruments.

(E) Propellers.

(F) Detail design.

(G) Materials and workmanship.

(H) Flying characteristics and qualities.

(I) Safety features.

Certain of these items may be demonstrated by analyses and drawings, others by visual inspection, and others by tests.

Sec. 4. Procedure Followed by the Department

(A) Drawings and stress analysis.--The drawings submitted by the applicant are examined for completeness, for conformity with what is considered good practice, and for conformity with the requirements stated herein. They are also checked against the stress analysis to ascertain whether or not the structure investigated conforms with that to be built. The stress analysis is checked for errors of assumption, for deviations from approved methods, and for errors in arithmetic.

(B) Application.--The application statement is examined for completeness and for conformity with the technical data.

(C) Structural tests.--When requested by the manufacturer, official witnesses for such structural tests as may be necessary will be provided. Manufacturer's affidavits, together with suitable photographs and test data, will be accepted in lieu of officially witnesses reports for minor tests.

(D) Inspection.--If the submitted design conforms with the requirements stated herein and is approved, the manufacturer is required to state under oath that the airplane which he submits for inspection is built in exact accordance with the approved specifications. An inspection is subsequently made to determine that the workmanship and materials incorporated in the airplane are such as to produce an airworthy machine. The empty weight of the airplane and the weight of each item of approved special equipment are exactly determined.

(*E*) *Flight tests.* — The airplane must then undergo the flight tests prescribed in chapter X. If such tests are successfully accomplished, the airplane will be approved. If, in addition, suitable manufacturing facilities are in evidence and the manufacturer so desires, an approved type certificate will be issued.

(*F*) *Approved data.* — When an approved type certificate is granted, one set of drawings is impressed with the seal of the Department of Commerce and returned to the manufacturer to be used in the construction of his airplane. The other set is placed in the files of the Department.

(*G*) *Conformity.*— Airplanes built to conform exactly to the approved drawings may, upon the manufacturer's making affidavit to that fact, be accepted as airworthy. Any deviations must first be approved by the Secretary of Commerce. Department of Commerce inspectors may call for, and must have access to, the sealed drawings when making an inspection at the manufacturer's plant to determine whether the airplanes as built conform to the approved data.

(*H*) *Validity.* — An approved type certificate is valid for as long a period of time as the manufacturer desires it to continue in force, provided that suitable manufacturing facilities are continuously maintained, and further provided that aircraft are being manufactured in accordance with the terms of the certificate. Manufacturing facilities are understood to include qualified personnel. The Secretary reserves the right to survey the outstanding approved type certificates periodically and to withdraw those for which such action is deemed advisable because of any of the reasons enumerated in section 5 of these regulations.

(*I*) *Reissue.*— Approved type certificates are nontransferable, but may be reissued to a manufacturer other than the original holder provided the former meets all the requirements pertaining thereto.

(*J*) *Semiannual report.* — A manufacturer to whom an approved type certificate has been issued shall file an affidavit with the Secretary on January 1 and July 1 of each year, stating the number of aircraft constructed in exact accordance with the terms of such certificate during the preceding 6 months. This report shall include the serial numbers or other identification marks for all the aircraft involved, together with the dates of manufacture and any other pertinent information.

(*K*) Technical data furnished by a manufacturer for approval will be treated as confidential.

Sec. 5 Revocation or Suspension of Approval

(*A*) Aircraft approved type certificates may be revoked or suspended for any of the following reasons:

(1) Violation, on the part of the manufacturer, of the Air Commerce Act or any regulations promulgated there under.

(2) Failure of the manufacturer to make proper and seasonable reports.

(3) Any false statement on the part of the manufacturer in application for approval or information accompanying the application, or in any report required under these regulations.

(4) Use or display of an approved type certificate or other official notification from the Department of Commerce for fraudulent purpose; or misrepresentation of any approved product.

(5) Use or display of an approved type certificate or other official notification from the Department of Commerce in any manner contrary to the public safety or interest.

(6) Any demonstration of incompetence, carelessness, or negligence, or the use of inferior or improper materials, on the part of the manufacturer.

(7) Failure of the manufacturer to maintain sufficient and suitable equipment and personnel to insure the airworthiness and exact adherence to approved specifications of the airplane manufactured.

(8) Refusal of the manufacturer to submit to inspection upon proper demand by a representative of the Secretary or to render any reasonable assistance in connection therewith.

(9) Moral irresponsibility of the manufacturer.

(*B*) Any unairworthy condition appearing in a particular aircraft type shall constitute grounds for revocation or suspension of prior approval of the type and of any airplane license previously issued to aircraft of the type in question.

Sec. 6. Drawings and Technical Data

(*A*) The applicant is required to submit an application for approval on a form furnished for this purpose by the Secretary of Commerce, a complete set of drawings, a stress analysis, and other pertinent data. The application shall be supported by an affidavit. The application and drawings shall be submitted in duplicate unless they are submitted to a branch office of the engineering section, in which case they shall be submitted in triplicate. One copy only of the stress analysis is required, except when it is submitted to a branch office of the engineering section, in which case two copies are required. Where a large number of drawings are involved, it may be convenient and will be acceptable to submit only one set of drawings for checking, together with duplicate drawing lists. The duplicate set or sets of drawings shall then be submitted upon completion of the checking.

(*B*) The data submitted in support of an application for approval shall include:

(1) *Drawings,* which shall be complete and well dimensioned and shall show the structure, power plant installation, seating arrangement and all other pertinent information in sufficient detail to be used for the construction and stress analysis of the aircraft involved. If the manufacturer so desires, adequate photographs may be substituted for drawings of the power plant installation, including cowling and exhaust system. Such photographs shall be made from marked negatives indicating the dimensions and materials of the piping and fittings. In any case a diagrammatic layout of the fuel and oil systems shall be submitted. The material used in each of the members of the primary structure, including fittings, shall be clearly indicated by specification number on the drawings. If heat-treated materials are used, the ultimate tensile strength and other means of positive identification shall be shown for each member. *Material specifications for all bolts, nuts, rivets, and similar standard parts used in the primary structure shall be definitely specified or referred to on the drawings.* All drawings shall bear a date of issue.

(2) *A balance diagram* showing the location of the centers of gravity of the component parts of the airplane and its useful load, the location of the mean aerodynamic chord, and the assumed center of pressure of the horizontal tail, together with table showing the weights of the items and the computations involved in determining the location of the composite center of gravity of the airplane with respect to the leading edge of the mean aerodynamic chord, for the following conditions:

(*a*) Airplane fully loaded.

(*b*) Center of gravity at most forward location for which approval is desired.

(*c*) Center of gravity at most rearward location for which approval is desired.

(3) *A list of all drawings submitted,* arranged numerically with respect to drawing number and, if desired, subdivided into groups, such as wing, fuselage, etc. The drawing list shall include all drawings originally submitted in connection with application for other models which also apply to the model in question without change. This list shall designate drawings by number, title, issue date, and airplane model for which originally issued. This list will be construed to make all the drawings enumerated on it a part of the data supporting the application for approval. When impressed with the seal of the Department, it will authorize the use of the construction show on those drawings in the model in question. This procedure may be modified, upon request, to agree with the manufacturer's filing system in cases where drawings covering all models are filed numerically in a general file. The data submitted shall also include *a list of standard equipment* which is furnished with the airplane. Such a list shall include brakes, starters,

landing lights, tools and similar items. There should also be submitted a list of such items as may be included as *special equipment,* together with the weight of each such item including the additional weight necessary for installation.

(4) *A stress analysis,* supplemented by test data if necessary, covering an investigation of all primary structural members for compliance with the requirements outlined in these regulations. Recommended procedure and methods are outlined in this bulletin and in Aeronautics Bulletin No. 26. The stress analysis shall state, by specification number, the material used for each member or group of members, whether or not it is heat-treated and what physical properties are guaranteed by the manufacturer. *The stress analysis shall also include a table indicating the margins of safety of all members. Buoyancy computations* shall be submitted for hulls and floats. The stress analysis shall bear the *signature* of the responsible engineer.

(*C*) For purposes of ready identification, the manufacturer shall designate each of his designs by a model number or letter. This number or letter shall be changed whenever any change is made in the design of such character that an application for a new type approval is necessary. All drawings shall show the number or letter of the original model to which they apply.

Sec. 7 Changes

(*A*) Changes in airplanes constructed under an approved type certificate are permissible, provided that they are approved by the Secretary.

(*B*) The suitability of minor changes is judged upon the basis of the airworthiness requirements which were in effect when the particular airplane model or type was originally approved, unless the specific circumstances indicate the advisability of compliance with current requirements. Minor changes which obviously do not impair the structural strength or reliability of the airplane nor affect its flying characteristics may be approved by engineering inspectors without prior reference to the Washington office. Shop drawings showing these changes shall be forwarded to the Department for record purposes as soon as possible after the changes are made.

(*C*) Major changes, such as the installation of an engine of a type other than that covered by the original airplane type approval, will require the issuance of a new type approval and may require compliance with current requirements at the discretion of the Department.

(*D*) When a revised drawing is submitted to the Department and all airplanes previously constructed according to the original drawing are not to be changed, it shall indicate the serial numbers of all airplanes to which the change applies. Corrected pages of the drawing lists should be submitted. Alternate installations should be so designated and properly indicated on the drawing lists.

Sec. 8. Structural Tests

(*A*) If members of unusual design are used, test data showing their strength properties under loads similar to those to which they will be subjected in the structure shall be submitted to substantiate the values assumed in the analysis.

(*B*) In particular cases where the structure cannot be satisfactorily analyzed and static tests are submitted as the only proof of compliance with the strength requirements, a strength test in accordance with section 11 (J) is required.

(*C*) The strength of wing ribs, fuel and oil tanks, control surfaces, and control systems shall be demonstrated by tests made on these units in the presence of a Department of Commerce representative. The loadings for these tests are outlined in the following chapters. When, in the opinion of the Secretary, the design of the landing gear is such as to warrant a drop test, such a test shall be made with the landing gear attached to the actual airplane structure, unless it is obvious that the result to be ascertained can be determined by the use of an approved jig in place of the actual airplane structure.

Sec. 9. Materials.

(*A*) The use of materials of inferior quality or of those which experience has shown to lack uniformity of quality or strength will be regarded as sufficient cause for withholding approval of a new design or for revoking approved type certificates or licenses already granted.

(*B*) The important physical properties of the materials used shall be definitely specified, either by reference to an accepted standard such as Army, Navy, or S.A.E., or by reference to reliable test results. Allowable loads and stresses shall be computed by means of standard structural formulas when possible. Army or Navy methods of determining allowable loads or stresses are, in general, acceptable. In the case of unconventional shapes or unusual stress combinations, reference shall be made to the source of the method used or to reliable test data. The effects of welding, form factors, stress concentration, and local failure shall be accounted for in the stress analysis. In the detailed design of members and fittings, care shall be taken toward preventing fatigue failures by proper material distribution and shape.

(*C*) The use of high-grade aluminum alloy, bronze, and steel castings will be permitted in members of the primary structure provided that the casting has a minimum ultimate strength of at least three times the critical applied load.

Sec. 10. Strength Requirements

(*A*) *The minimum factor of safety for any aircraft structure or component thereof shall be 1.50 unless otherwise specified.* — This requires that the ultimate strength of any member shall be at least 1.50 times as great as its critical applied load. (See sec. 11 (B).)

(*B*) All allowable loads or stresses, whether determined from computations or tests, shall be based on the standard *guaranteed* physical properties of the material.

(*C*) Negative margins of safety are not acceptable unless it is conclusively shown that the apparent deficiency in strength is offset by conservative design features. (See sec. 11 (E).)

(*D*) The applied stress in any material shall not exceed the yield point. In general, the minimum factor of safety specified in paragraph (A) of this section is sufficiently high to provide for this.

(*E*) For materials which have no well-defined yield point, the yield point shall be defined as that stress which produces a permanent set of 0.002 inch per inch length. The minimum factor of safety specified in paragraph (A) of this section will usually insure compliance with this requirement, except in certain cases such as aluminum alloy members carrying critical tension of bending loads.

Sec. 11. Definitions

(*A*) *The applied load factor is,* in general, the actual acceleration in terms of g , the acceleration of gravity, expected to be applied in a given flying or landing condition. It represents the ratio between the total loads in the accelerated and unaccelerated conditions. An applied load factor of n acting in a given direction therefore indicates that the actual total external load acting on the airplane in that direction is n times the weight of the airplane. In such an accelerated condition (with no angular acceleration) each item of mass in the airplane exerts a force on the structure equal to n times its own weight, and in a direction parallel to and opposite to that of the external load.

(*B*) The *applied loads* are the actual loads, either external or internal, produced by a specified loading condition. External applied loads are usually expressed as unit loading in pounds per square foot in the flying conditions. The internal applied loads may be expressed directly in pounds or converted into applied shear, bending or torsional moment, or unit stress.

(*C*) The *design loads* are equal to the applied loads multiplied by the total factor of safety.

(*D*) The *factor of safety* is an arbitrary factor by which the applied loads or load factors are multiplied for the purpose of insuring sufficient strength to permit the applied loads to be exceeded by a definite amount before complete failure of the structure occurs. In general, the factor of safety also provides sufficient strength to prevent permanent set under the applied loads. The total factor of safety may include special factors to insure extra strength or rigidity in certain cases.

(*E*) The *margin of safety* is the percentage by which the *ultimate strength* of a member exceeds the design load for the member. (See sec. 10 (C).)

(*F*) A *linear margin of safety* is one which varies linearly with the total design load.

(*G*) A *nonlinear margin of safety* is one which is based on stresses which are not proportional to the total design load. A nonlinear margin of safety is not a true measure of the excess strength of a member.

(*H*) The *design speed* is the airplane speed at which the specified design condition is assumed to occur. For stress analysis purposes the "indicated" air speed is used. The "indicated" air speed is defined as the speed which would be indicated by a perfect air-speed indicator, namely, one which would indicate true air-speed at sea level under standard atmospheric conditions. See Aeronautics Bulletin No. 26, sec. 6 (*A*) (8), for further information on this subject.

(*I*) A *proof test* is a test in which the structure is subjected to the *applied* load, properly distributed, for a period of at least 1 minute. In a proof test no permanent set is permissible. In determining permanent set the effects of slippage or jig deflection may be deducted if properly measured.

(*J*) A *strength test* is used for determining the ability of a structure to withstand its *design load* , properly distributed. To cover material variations, strength tests should be carried to 125 percent of the design load, unless the test results are properly corrected to account for the variation of the actual properties of the material in critical portions of the test structure from the standard guaranteed properties. The pertinent actual properties of the material shall be obtained by making tests of specimens taken from the material used in the test structure. Permanent set is permissible in a strength test, provided that the structure supports the required loading before failure. Strength tests may be conducted in lieu of a stress analysis, provided that the test program is submitted for approval before the test is conducted.

Chapter II. — BASIC FLYING CONDITIONS

Sec. 12. Terms and Coefficients Used in Flying Conditions

(*A*) *Effective wing area.* [1]— In investigating the basic flying conditions for stress analysis purposes, the effective area shall be considered as the actual projected area of the wing on the plane of the chords, including the aileron area, with the following modifications:

[1] In order to simplify computations, the plan form of tapered or elliptical wings may be represented by a number of trapezoids closely approximating the actual plan form and having an equivalent area.

(1) That portion of the wing over which either or both surfaces are replaced by the fuselage shall be omitted, unless the actual span distribution is determined from wind-tunnel tests covering a suitable range of angles of attack. Lift produced directly by the fuselage shall be neglected. The maneuvering loads are considered to be produced entirely by the wing.

(2) Trailing edge cut-outs may be neglected if they do not remove more than one-half the chord.

(3) The effect of nacelles may, in general, be neglected in computing effective area.

(*B*) *Effective wing loading.*— The effective wing loading is equal to the gross weight of the airplane divided by the total effective area.

(*C*) *Power loading.* — The power loading is equal to the gross weight of the airplane divided by the total rated horsepower.

(*D*) *Aerodynamic coefficients.* — The absolute (nondimensional) coefficients used in the design conditions refer to the effective wing area as defined in paragraph (A) of this section. (Average unit loadings in either the lift, drag, or chord directions are all found by dividing the proper component of the total load by the same wing area.)

(*E*) *Normal force coefficient.*— The normal force coefficient is the ratio between the applied average unit air load acting normal to the plane of the wing chord and the dynamic pressure corresponding to the design speed. For a given flight condition the applied unit air load, in pounds per square foot, is therefore proportional to the force coefficient.

(*F*) *Chord force coefficient.* — The chord force coefficient corresponds to the normal force coefficient except that it refers to the component of the applied average unit air load acting in the plane of the wing chord.

(*G*) *Standard coefficients.* — The lift, drag, and moment coefficients have their usual significance.

(*H*) In general, the normal and chord force coefficients as referred to the basic wing chord (used in specifying the aerodynamic characteristics of the airfoil) may be used directly for stress analysis purposes. When the mean plane of the drag truss is not parallel to the plane of the basic chords or not perpendicular to the plane of the spars, corrected coefficients should be determined by properly resolving the resultant force coefficient into the equivalent coefficients acting in the respective planes of the beams and drag truss. Corrections for aspect ratio and taper in plan form should also be made. (See Aeronautics Bulletin No. 26, secs. 5, 7, and 9.)

Sec. 13. General Design Conditions

(*A*) The accelerated flying conditions will be based on two design airplane speeds. The applied load factors to be used with these speeds will be specified, together with certain arbitrary requirements pertaining to aerodynamic characteristics. The flying conditions specified for the design of the wings are considered to represent the minimum amount of investigation necessary to insure sufficient strength for all normal maneuvers and for flying through gusts.

(*B*) The applied load factors specified for maneuvering conditions are based largely on past experience and are therefore of a semi-empirical nature. It should be realized that these load factors are minimum factors which might easily be exceeded in severe maneuvering. The Department will consider, as a substitute for the specified values, a rational determination of the maximum acceleration likely to be obtained in maneuvering, based on reliable wind-tunnel or flight-test data.

(*C*) The gust load factors are obtained from simplified formulas which neglect the effects of the gust gradient, the increase in the resultant velocity of the air, flexibility of the wing structure, and similar secondary features. The gusts are considered as acting normal to the flight path. It should be noted that a decrease in airplane gross weight will increase the gust load factor. This may cause critical loads to be developed in parts of the structure supporting dead weight and should therefore be considered in connection with certain types of airplanes having a widely variable loading.

(*D*) The lift or normal force coefficient obtained in the gust conditions may exceed the maximum lift coefficient determined from windtunnel tests. When this occurs the computed value shall be used and other characteristic curves should be conformably extended as outlined in Aeronautics Bulletin No. 26, sec. 7 (C).

Sec. 14. Design Speeds

(*A*) The two basic design speeds are designated VL and Vg. VL denotes the high speed of the airplane in level flight, reduced to the equivalent "indicated" air speed at sea level in standard air, as defined in section 11 (H). (See also Aeronautics Bulletin No. 26, sec. 6.) Vg represents the design gliding speed used in the analysis of the structure and is also considered to be "indicated" air speed, as in the case of VL.

(*B*) The high speed, VL, should be estimated as accurately as possible or determined from flight tests. The design high speed used in the stress analysis shall not be less than the high speed of the airplane finally determined from flight tests and reduced to the equivalent "indicated" air speed as defined in section 11 (H).

(*C*) The minimum design gliding speed shall be determined from the following equation:

Vg = VL + Kg (Vm - VL) (need not exceed V_L + 150 ft./sec.)

Where V_L = design high speed. (See par. (B) of this section.)

Vm = maximum theoretical vertical speed in standard air with zero propeller thrust.

$K_g = 0.08 + \frac{1850}{W + 3000}$ (shall not be less than 0.15).

W = airplane gross weight in pounds.

(1) In the above equation the term Vm is used to introduce the effects of low airplane drag on gliding speed and should not be considered as a design speed. (See Aeronautics Bulletin No. 26, sec. 6, for method of computation.)

(2) The minimum design gliding speeds as determined by the above method correspond, in general, to relatively flat gliding angles with zero thrust. For airplanes which are to be dived to high speeds the manufacturer is responsible for the choice of a suitable design speed. The use of the actual terminal velocity is recommended for such cases.

(3) In accordance with section 64 (H), the airplane will be placarded for a limited speed 10 miles per hour less than the design gliding speed used in the stress analysis and must meet all stability requirements without exceeding the placarded limited speed. This consideration may necessitate the choice of a design gliding speed for stress analysis purposes somewhat greater than the minimum required.

Sec. 15. Basic Accelerated Flying Conditions[1]

[1] See sec. 11, Aeronautics Bulletin No. 26.

(*A*) The following symbols will be used in outlining the basic accelerated flying conditions:

V = design speed, feet per second (= 1.47 times speed in miles per hour.)

n = applied wing load factor (normal to basic wing chord).

W = gross weight in pounds.

s = effective wing loading. (See sec. 12 (B).)

p = power loading. (See sec. 12 (C).)

C_M, C_C, etc. = actual values for airfoil section used.

C_M, C_C, etc. = modified values specified for design.

R = aspect ratio. (See Aeronautics Bulletin No. 26, sec. 7.)

m_6 = slope of lift curve, ΔC_L per radian, at R = 6.

q= dynamic pressure = $\frac{\rho}{2} V^2$.

$q_L = 11.9 \left(\frac{V_1}{100}\right)^2$ (V is in feet per second).

$q_L = 11.9 \left(\frac{V_g}{100}\right)^2$ (V is in feet per second). (See sec. 14 for design speeds.)

(*B*) *Condition I.* — Positive high angle of attack.

(1) $n_1 = 1.0 + \Delta n_1$ (shall not be less than 2.50).
Where Δn_1 is the higher value determined from the following equations:

(a) $\Delta n_{1a} = 0.036 \frac{V_1 m_6}{s} \left(\frac{4}{3 ÷ 6/R}\right)$

(b) $\Delta n_{1b} = \left[0.77 ÷ \frac{3200}{W ÷ 9200}\right] K_1$

(See fig. 1 for value of K_1.)

For float type seaplanes and for amphibians Δn as determined from (b) may be reduced 5 percent.

(2) $C_{n1} = n_1 s / q_L$.

(3) Cc'= value corresponding to C_{N1}, or = -0.20 C_{N1}, whichever is greater. (See Aeronautics Bulletin No. 26, sec. 3.)

(4) C.P.'= most forward position between $C_L = C_{N1}$, and C_L max.

(When C_{N1} approaches or exceeds C_L max. the C.P. curve should be extended accordingly. (See Aeronautics Bulletin
No. 26, sec. 7 (C).)

(*a*) For biplane combinations the center of pressure of the upper wing shall be assumed to be 2.5 percent forward of its nominal position.

(5) CM’= moment coefficient necessary to give required C.P.' in conjunction with C_{N1}. (See Aeronautics Bulletin No. 26, sec. 3, for equation.)

(*C*) *Condition II*. — Negative high angle of attack.

(1) $n_2 = 1.0 - \Delta n_{1a}$.

Where Δn_{1a} is obtained from paragraph B (1) (a).

(2) $C_{N2} = n_2 s/q_L$.

(3) C_c = actual value corresponding to C_{N2}.

(*a*) When C_c is positive or has a negative value less than 0.02, it may be assumed to be zero.

(4) C_M = actual value corresponding to C_{N2}.

(*D*) *Condition III* . - Positive low angle of attack.

(1) $n_3 = 1.0 + \Delta n_3$ (shall not be less than 2.0).

Where Δn_3 is the higher value determined from the following equations:

(a) $\Delta n_{3a} = 0.018 \frac{V_L m_s}{s}\left(\frac{4}{3 \div 6/R}\right)$

(*b*) $\Delta n_{3b} = 0.60\, \Delta n_{1b}$ (as determined in par. B (1) (b).)

(2) $C_{n3} = n_3 s/q_g$.

(3) C_c = actual value corresponding to C_{n3}.

(*a*) When C_c is positive or has a negative value less than 0.02, it may be assumed to be zero.

(4) $C_M' = C_M - 0.01$, where C_M is the actual value corresponding to C_{N3}.

(*a*) The increment — 0.01 is added to account for roughness, inaccuracies in rib construction, distortion, and improperly rigged ailerons or flaps. It will be waived in cases where such effects are eliminated or reduced to a negligible amount.

(*E*) Condition IV. — Negative low angle of attack.

(1) $n_4 = 1.0 - \Delta n_{3a}$.

Where Δn_{3a} is obtained from paragraph D (1) (*a*).

(2) $C_{N4} = n_4 s/q_g$.

(3) C_c = actual value corresponding to C_{N4}.

(*a*) When C_c is positive or has a negative value less than 0.02 it may be assumed to be zero.

(4) $C_M' = C_M - 0.01$, where C_M is the actual value corresponding to C_{N4}.

(*a*) Paragraph D (4) (a) also applies in this case.

Sec. 16. Balancing the Airplane

(*A*) The external applied loads required to hold the airplane in an assumed state of equilibrium shall be determined for the four basic accelerated conditions outlined in section 15, using VL for conditions I and II and Vg for conditions III and IV. Balancing loads shall also be determined for the design conditions required when wing flaps are used. See sec. 18 (D). Simplified methods for determining the balancing loads are outlined in Aeronautics Bulletin No. 26, sec. 12.

(*B*) When the full-load center of gravity position is variable (as in large transport airplanes) the airplane shall be balanced for both extreme positions unless it is apparent that only one is critical. In certain cases it may also be necessary to check the balancing tail loads required for the loading conditions which produce the most forward and most rearward center of gravity positions for which approval is desired.

Chapter III. WINGS AND WING BRACING[1]

[1] Includes cabane bracing.

Sec. 17. Design Conditions

(*A*) The wings and wing bracing shall be investigated for the accelerated flying conditions I, II, III, and IV, as outlined in section 15. To cover special conditions and to insure suitable structural stiffness, the modifications, supplementary conditions, and special requirements outlined in the succeeding sections shall also be complied with. These modifications and supplementary conditions apply only to the wings and wing bracing unless otherwise noted.

Sec. 18. Modifications to Basic Accelerated Conditions[2]

[2] See sec. 14, Aeronautics Bulletin No. 26.

(*A*) *Condition* I_1. —

(1) The smaller of the two values of Cc specified in section 15 (B) (3) and the most rearward C.P. position in the range specified in section 15 (B) (4) shall also be investigated when Condition I is critical for the rear spar (or its equivalent), or if any portion of the front spar (or its equivalent) is likely to be critical in tension.

(*B*) *Condition* III_1.—

(1) The effects of displaced ailerons on the moment coefficient shall also be considered for that portion of the span incorporating ailerons, unless the basic moment coefficient of the airfoil at zero lift has a negative value equal to or greater than 0.05. In general, it is satisfactory to assume that the effect is to increase the basic value of C_M to — 0.05.

(*C*) *Condition* IV. —

(1) The effect of displaced ailerons need not be investigated for this condition.

(*D*) *Effects of trailing-edge flaps.—*

(1) When wings are equipped with trailing edge flaps, the moment coefficient over that portion of the span affected by the flaps will be increased negatively for a downward displacement of the flaps. (This corresponds to a rearward displacement of the center of pressure.) This should be considered in the analysis of conditions II, III, and IV, but will not usually need to be considered in condition I. In condition III with displaced flaps the applied load factor shall be computed from section 15, paragraph D (1) (a), as the equation specified in paragraph D (1) (b) does not apply to the displaced flap condition. The airfoil characteristics and flap characteristics should be determined from reliable test data such as are contained in N.A.C.A. Reports and Technical Notes.

(2) When automatic devices are employed which positively prevent the flaps from opening or remaining open at speeds above a certain predetermined speed, the latter may be used in the analysis instead of the specified design speeds.

(3) When flaps of relatively large area are employed, a suitable modified design speed may be used instead of the specified design speeds and only conditions III and IV need be investigated as outlined in paragraph (1), provided that the following requirements are complied with:

(*a*) The design speed used for the extended flap conditions shall not be less than twice the minimum flying speed with flaps extended.

(*b*) The placard specifying the limited gliding speed for the airplane will be amplified as specified in section 64, paragraph (I).

(*c*) The analysis shall be made for the most critical flap position, which will generally be the fully-extended position.

(*d*) The flap operating mechanism shall be such as to minimize the chances of sudden, inadvertent, or automatic opening of the flap at speeds above the design speed for the extended flap conditions.

(*e*) When partial-span flaps are used, the span loading shall be suitably modified.[1]

> [1] Pending further investigation of the effects of flaps on span loading, the effect of a flap on the airfoil characteristics may be assumed to be confined to that portion of the wing incorporating the flap.

(*f*) When any type of split flap is employed, the inherently high drag (hence rearward chord coefficient) necessitates an investigation of the supplementary wing design condition outlined in sec. 19 (B). This requirement may be met by suitably modifying the chord coefficient used in condition III, when it is conservative to do so.

(4) Flaps shall be installed as to eliminate any tendency to induce wing flutter. This requires especially rigid attachment members and control system. An oleo damping device is recommended for high-speed airplanes having cantilever wings.

(5) The design conditions for the flap structure and control system are outlined in chapter IV.

(*E*) *Effects of slots and auxiliary airfoils. —*

(1) In the design and analysis of wings incorporating slots or auxiliary airfoils, the same general principles as those outlined in paragraph (D) of this section will apply, except that the use of a modified design speed is limited to cases

in which the auxiliary device can be retracted into or made an integral part of the wing proper. The individual aerodynamic characteristics of the auxiliary devices as well as the modified characteristics of the basic airfoil should be determined from reliable test data. Preliminary design data shall be submitted in all cases where unconventional auxiliary devices are employed.

Sec. 19. Supplementary Wing Design Conditions[2]

[2]See sec. 15, Aeronautics Bulletin No. 16

(*A*) *Condition V.* — Inverted flight.

(1) For *externally braced wing structures,* an approximate determination of the loads in the rear lift truss system (or in a single lift truss system) shall be made under the following assumptions:

(*a*) $n_5 = -1.0 - \Delta n_5$ (minimum negative value = 1.50)

Where Δn_5 is the higher of the two following values:

$\Delta n_5 = 0.50 \Delta n_{1a}$ (where n_{1a} is determined from sec. 15 (B) (1) (*a*)).

$\Delta n_{5b} = 0.25 \Delta n_{1b}$ (where n_{1b} is determined from sec. 15 (B) (1) (*b*)).

(*b*) $C_c' = 0$

(*c*) C.P.' = 25%, or, $C_{ma} = 0$.

(*B*) *Condition* VI. —*Special diving condition* .

(1) To insure sufficient strength in the drag truss system the drag trusses of each wing, or their equivalent, shall be investigated for the following condition:

(*a*) $V = V_g$.

(*b*) $C_N = O$.

(*c*) $C_c' = C_c$ max. (positive) + 0.01.

Note. — The increment of 0.01 is added to account for surface roughness and protuberances. It may be omitted when such effects are eliminated by providing exceptionally clean and smooth wing surfaces.

(2) In certain types of structures (such as biplanes and skin stressed wings) in which the loads due to wing moment and those due to rearward chord forces combine to give critical loads, the actual conditions at Cc max. should be investigated as follows:

(*a*) $V = V_g$.

(*b*) C_N = value corresponding to Cc max. (positive).

(*c*) $C_c' = C_c$ max. (positive) + 0.01.

(d) $C_M' = C_M - 0.01$.

Where C_M is the actual value corresponding to C_N (from par. (b)). (See sec. 15 (D) (4) (a) and 18 (B) (1).)

(3) In either of the above conditions, conservative assumptions covering the drag of the nacelles and similar items shall be made.

(*C*) *Unsymmetrical flying conditions.* —

(1) The wing bracing, wing attachment fittings, and the fuselage structure adjacent to the wing shall be investigated for the unsymmetrical loading produced by the use of the ailerons or by encountering uneven gusts. The following arbitrary rules (par. (2), (3), and (4)) may be used pending the development of a more rational method.

(2) Conditions I and III shall be investigated, assuming 100 percent of the applied air load to be acting on one side of the airplane and 70 percent on the other. For airplanes over 10,000 pounds gross weight the latter factor may be increased linearly with gross weight up to 80 percent at 25,000 pounds.

(*a*) When condition III_1 applies, the loading for this condition should be used on the 100 percent side, the loading for condition III being used on the opposite side.

(3) In the unsymmetrical flying conditions the angular inertia of the wings shall be neglected, except that the effect of nacelles may be considered.

Note. — Special consideration will be given to cases in which this arbitrary ruling appears to be unduly conservative.

(4) The unsymmetrical condition should also be applied to the supplementary inverted flight condition outlined in section 19 (A), in some types of construction, particularly those employing cabane bracing.

Sec. 20. Load Distribution

(*A*) *Biplane distribution.*— The distribution of air load between the wings of a biplane shall be determined in accordance with the best available method or from reliable wind-tunnel tests. The moment coefficient shall be assumed to be unaffected except as modified in condition I under center of pressure requirements. Acceptable procedure is outlined in Aeronautics Bulletin No. 26, sec. 7 (D).

(*B*) *Spanwise distribution.*[1]—

[1]A standard rational method of span distribution will be adopted when sufficient data are available.

(1) The normal force coefficient (C_N) shall be assumed to vary along the span in accordance with figures 2 and 3, which are assumed to represent the two extreme cases of tip loading. Each case should be investigated, unless it is demonstrated that only one is critical. In connection with figure 3, the wing tip is defined as the wing area outboard of the station at which the wing chord and distance to the extreme tip are equal. A more exact span distribution may be used if adequate test data are available. (See sec. 12 (A).) A single separate span distribution may be used for each design condition if the actual distribution for the design value of C_N is known, including the effects of cut-outs and fuselage interference.

(2) The effects of nacelles on the normal force coefficient may, in general, be neglected. Their effects on chord loads are outlined in section 21.

(3) The effect of trailing edge cut-outs which remove less than 50 percent of the chord may be neglected when figures 2 and 3 are used.

(4) When the normal force coefficient is assumed to vary over the span, the values used shall be adjusted so as to give the same total normal force as the design value of C_N acting uniformly over the span. (See Aeronautics Bulletin No. 26, sec. 9, for additional information.)

(5) The chord coefficient shall be assumed to be constant along the span, that is, it is assumed that tip loss does not affect the chord coefficient. The effects of wing flaps should be considered, however.

(6) The center of pressure or the moment coefficient (whichever is specified) may be assumed to be constant over the span, except as affected by ailerons or flaps.

(*C*) *Chord distribution.*— Wing ribs.

(1) The loadings specified in the succeeding paragraphs are satisfactory for the usual type of wing rib, comprising a complete truss between leading and trailing edges. An investigation of the actual chord distribution should be made in unconventional cases such as single spar wings, or skin-stressed wings in which longitudinal stiffeners are also used to support direct air loads. Information on chord distribution is contained in Aeronautics Bulletin No. 26, sec. 22.

(2) The strength and dependability of wing ribs will be judged from the drawings, together with the results of *tests to destruction.* Analyses of ribs will in no case be accepted as a satisfactory means for demonstrating their strength. The test set-up shall simulate the actual conditions on the airplane. Wood ribs shall not be glued to the test spars.

(3) Wing ribs shall be tested for both a positive high angle of attack condition (condition I) and a medium angle of attack condition. The total design load to be carried by each rib shall equal the design normal load over the area supported by the rib, modified as specified in paragraphs (7) and (8). For the medium angle of attack condition, the load factor shall be taken as the average of the *design* load factors for conditions I and III.

(4) For the high angle of attack condition ribs having a chord length greater than 60 inches shall be subjected to 16 equal loads so arranged as to be applied at 1.0, 3.0, 5.0, 7.3, 9.9, 12.9, 16.2, 19.9, 24.1, 28.9, 34.2, 40.4, 47.5, 56.5, 72.0, and 90.0 percent of the chord. The sum of these loads shall equal the total load carried by the rib, computed as specified in paragraph (3) of this section. For ribs having a chord of less than 60 inches, 8 equal loads may be used, their arrangement being such as to produce shears and moments of the same magnitude as would be produced by the application of 16 equal loads at the locations specified above.

(5) For the medium angle of attack condition 16 equal loads shall be used on ribs of chord greater than 60 inches, 8 equal loads for chords less than 60 inches. In either case the total load shall be computed as specified in paragraph (3) of this section, for the medium angle of attack condition. When 16 loads are used, they shall be applied at 8.34, 15.22, 19.74, 23.36, 26.60, 29.86, 33.28, 36.90, 40.72, 44.76, 49.22, 54.08, 59.50, 65.80, 73.54, and 85.70 percent of the chord. When 8 loads are used they shall be so arranged as to give comparable results.

(6) No less than two ribs shall be tested in either loading condition. For tapered wings a sufficient number of ribs shall be tested to show that all ribs are satisfactory.

(7) On ribs where the lacing cord for attaching the fabric passes entirely around the rib, all of the load shall be applied on the bottom chord. Such ribs shall support a total load of 125 percent of the design load.

(8) On ribs where the covering is to be attached separately to the two chords of the rib, the loading specified in paragraph (3) of this section shall be modified so that approximately 75 percent of the total comes on the top chord and 50 percent on the bottom, the total load being 125 percent of the design load.

(9) The leading edge portion of the rib may be very severely loaded in condition II or IV. An investigation of the maximum down loads on this portion shall be made when the high speed of the airplane exceeds 150 miles per hour. (See Aeronautics Bulletin No. 26, sec. 22 (B)). Where this requirement does not apply, it shall be demonstrated that the rib structure ahead of the front spar is strong enough to withstand its portion of the test load specified in paragraph (4) of this section, acting in the reverse direction. A test for this condition will be required in the case of a rib which appears to be weak.

Sec. 21. Factors Influencing Wing Loads

(*A*) *Dead weight.* — Each item of weight in or attached to the wing cellule, including the weight of the structure itself, shall be considered in the determination of the net forces acting on the structure. When a considerable amount of dead weight is supported by the wing structure, *the wing and wing bracing shall also be analyzed for the landing conditions.* When fuel tanks of considerable capacity are supported by the wing structure, the investigation should cover both full and empty tanks.

(*B*) *Parasite drag.* — The drag of large items attached to the wing cellule (such as nacelles) shall be estimated and considered in conjunction with the conditions in which the addition of such a drag load increases the loads in the critical members.

(*C*) *Propeller thrust*.— The propeller thrust from nacelles may be neglected in the detailed analysis of the wing structure, with the following exceptions:

(1) When the nacelle location is such as to produce large local loads on the wing structure (nacelle above wing, etc.).

(2) When, in multiengined airplanes, nacelles are located at a considerable distance from the plan of symmetry, in which case the wing attachment structure should be analyzed for the case of full power applied on one side only.

Sec. 22. Requirements for Lift Trusses

(*A*) When streamline wires are used for lift wires, they shall be double unless the wings are so designed that with any lift wire removed the strength of the remaining wing structure will be adequate to sustain 50 percent of the required load factors. When parallel double wires are used the design load computed for each wire shall be increased 5 percent, except in cases where the necessary increase is covered by an additional factor of safety to reduce deflection.

(*B*) When two or more wires are attached to a common point on the wing but are not parallel, the approximate method outlined in Aeronautics Bulletin No. 26, sec. 23 (B), may be employed in lieu of a more exact method, provided that the loads so computed are increased 25 percent. These increased loads shall be carried through the structural members to which the wires are attached.

(*C*) In lift trusses which are braced by wires only, the wire sizes and factors of safety shall be determined in accordance with section 62.

(*D*) All members of a wire-braced lift truss shall be designed to provide the minimum factor of safety of 1.5 against the *applied* rigging loads computed in accordance with section 62 (C). This applies also to drag truss members which are directly loaded by rigging the lift truss wires, in which case the combined effects of rigging the drag truss and lift truss wires shall be investigated.

(*E*) External wing-brace struts which are at an angle of more than 45° with the plane of symmetry and which have a cross-sectional fineness ratio of more than 3 shall be assumed to act as lifting airfoils and shall be designed to carry the resultant transverse loads in combination with the specified design axial loads. For this purpose the strut sections shall be assumed to have a normal force coefficient equal to 1.0 and the total air load shall be based on the exposed area of the strut. The chord components and vertical reactions of this air load and the lift contributed by the strut shall not be considered in the analysis of the wing.

Note. — A rational determination of the actual force coefficient for the strut will be accepted, if sufficiently conservative.

(*F*) On monoplane wings which are externally braced by wires only, the right and left sides of the bracing shall be independent of each other so that an unsymmetrical load from one side will not be carried through the opposite wires before being counteracted. This ruling will be waived when the following conditions are complied with:

(1) The minimum true angle between any external brace wire and a spar shall be approximately 14°.

(2) The counter (landing) wires shall be designed to remain in tension up to the applied load or somewhat beyond.

(3) Additional special conditions representing the most severe unsymmetrical flying or landing loads shall be investigated.

(4) Both landing and flying wires shall be double.

(*G*) Wing cellulose in which the division of loading between lift trusses and drag trusses is indeterminate shall be analyzed either by an acceptable method for indeterminate structures or by making assumptions which result in conservative design loads for all members. This applies particularly to the distribution of wing moment in the diving condition. (See Aeronautics Bulletin No. 26, sec. 23 (C))

Sec. 23. Requirements for Drag Trusses

(*A*) Fabric-covered wing structures having a cantilever length whose ratio of span to mean chord is greater than 1.75 shall have a double system of internal drag trussing spaced as far apart as possible. The design loads for the counter wires shall be the same as for the drag wires. The minimum factor of safety shall be multiplied by an additional factor as specified in paragraph (B).

(*B*) Whenever double drag trussing is employed, all drag wires or their equivalent shall have an additional factor of safety varying linearly from 3.0 when the ratio of overhang to mean chord is 2.0 or greater to 1.20 when the ratio is 1.0 or less, assuming an equal division of drag load between the two systems.

(*C*) The drag-wire sizes shall be so proportioned as to comply with the specifications of section 62, paragraphs (A), (B), and (C).

(*D*) Drag struts shall be assumed to have an end fixity coefficient of 1.0 except in cases of unusually rigid restraint, in which case a coefficient of 1.5 may be used. Drag struts in double-truss systems shall be designed to withstand the maximum shear developed when the drag wire of the upper system in one bay and the drag wire of the lower system in the adjacent bay are each carrying their design loads, the remaining wires in these two bays being assumed to be out of action.

Sec. 24. Requirements for Spars

Recommended methods of analysis and design for spars are outlined in Aeronautics Bulletin No. 26, secs. 16 to 20. The following requirements shall be met in all cases:

(*A*) In the analysis of wing spars subjected to combined axial and transverse loading the effects of secondary bending shall be accounted for. In such cases the required factor of safety shall be introduced *before* the computation of the bending moments in order to insure that the design loads can be supported by the structure.

(*B*) Provision shall be made to reenforce the spars against torsional failure, especially at the point of attachment of lift struts, brace wires and aileron hinge brackets.

(*C*) In all beams, whether of wood or metal, an investigation shall be made of the strength of the beam against lateral buckling.

(*D*) The ability of box spars to resist longitudinal shear in the webs shall be demonstrated by analysis or tests.

(*E*) The use of a beam formula for the computation of truss deflections will not be approved, except where tests have been made of the actual truss to determine its effective EI.

(*F*) Where a joint in a wood spar is designed to transmit bending from one section of the spar to another or to the fuselage, the stresses in each part of the structure shall be calculated on the assumption that the joint is 100 percent efficient and also under the assumption that the bending moment transmitted by the joint is 75 percent of that obtained under the assumption of perfect continuity. Each part of the structure shall be designed to carry the most severe load determined from the above assumptions. In the analysis of metal spars, perfect continuity may be assumed and shall be maintained in the actual structure by suitable design.

(*G*) When a spar joint is at or near the mid-point of a bay, it shall be of such design as to maintain the continuity of the spar and need not be analyzed for the condition of 75 percent continuity. In such cases tests may be required to demonstrate that the joint does not increase the deflection at that point.

(*H*) In computing the area, moment of inertia, etc., of wood spars pierced by bolts the diameter of the bolt hole shall be assumed to be 25 percent greater than the diameter of the bolt, except that the assumed diameter of the hole need not be more than one-eighth inch in excess of the bolt diameter.

(*I*) In computing the properties of the box spars for final design, only that portion of the web with its grain parallel to the spar axis and one-half of that portion of the web with its grain at an angle of 45° to the spar shall be considered. If desired, the more conservative method of neglecting the web entirely may be employed.

Sec. 25. Special Requirements

(*A*) For the design of the members attaching the ailerons to the wing structure an additional multiplying factor of safety of 1.5 shall be employed. If the ailerons are statically or dynamically balanced, this requirement will be waived.

(*B*) The use of slots, auxiliary airfoils, or unconventional ailerons may require additional investigation and modification of the basic requirements. Preliminary data should be submitted in such cases outlining the proposed methods of analyses.

(*C*) In the fabric-covered wings, ordinary wing ribs shall not be used to support the ailerons or aileron false spars. Separate attachment brackets or reinforced ribs are required.

(*D*) Internally braced biplanes shall be provided with N or I struts to equalize deflections, and the effect of such struts shall be considered in the stress analysis.

(*E*) Multiple-strand cable shall not be used in the usual types of lift trusses or drag trusses. (Exceptions to this rule may be made for certain types of airplanes.)

(*F*) All structural members in the wing lift truss system which transmit direct loads from the landing gear shall be overstrength with respect to landing loads, so as to reduce the probability of damaging the wing in severe landings. This will usually require that, for any given landing condition, the minimum margin of safety in the lift truss structure shall be at least 10 percent greater than the minimum margin of safety in the landing gear structure for the same condition.

(*G*) Fabric-covered cantilever wings shall be tested to determine their torsional stiffness. Such a test may be conducted by applying a pure torsional couple near the wing tip and measuring the resulting angular deflection at the point of application.

(*H*) The strength of multi-spar, plywood-covered, and metal-covered wings shall be substantiated by tests or by stress-analysis methods which conservatively account for redundancies and local failures. Care should be taken to prevent the occurrence of wrinkling or buckling of metal wing coverings at relatively low applied load factors. Information on unconventional types of wing structure is contained in Aeronautics Bulletin No. 26, sec. 21.

Chapter IV. CONTROL SURFACES AND AUXILIARY DEVICES[1]

Sec. 26. Horizontal Tail Surfaces[2]

1 See sec. 21. Aeronautics Bulletin No. 26, ch. IV.
2 See sec. 29(b) for effects of trailing edge tabs.

(*A*) *Balancing loads.*—

(1) The *applied* load for the design of horizontal tail surfaces shall not be less than the maximum balancing load determined from conditions III and IV (sec. 15, paragraphs (D) and (E)). This also applies to the special design conditions used for displaced wing flaps (see sec. 18 (D)).

(2) In computing the required load over the fixed portion of the tail surface (stabilizer) it shall be assumed that an oppositely directed load is applied from the elevator at the elevator hinge-line. This load shall equal 40 percent of the required net applied tail load specified in paragraph (1), except that it shall not be less than that corresponding to the minimum control force specified in sec. 31 (A) (2) and need not be greater than that corresponding to a maximum control force of 150 pounds, based on the distribution illustrated in Fig. 5.

(Note. — Special consideration will be given to cases in which it can be shown that the opposite elevator load determined by the above method is impossible to obtain under any loading condition.)

(3) The distribution over the stabilizer chord shall be in accordance with figure 4. The unit loading may be assumed to be constant over the span in all cases.

(*B*) *Maneuvering loads.*—

(1) Horizontal tail surfaces shall also be designed for *applied* unit loadings not less than those corresponding to a pull-up speed V_p and normal force coefficients of -0.55 and +0.35 (+ upward, -downward.)

(*a*) $V_p = V_s + K_p(V_L - V_s)$.

Where V_s = minimum speed of level flight (stalling speed).

V_L = maximum speed of level flight. (See sec. 14.)

(shall not be less than 0.5).

W = gross weight of airplane in pounds.

(*b*) .

Where = average applied unit load over entire horizontal surface.

q_p = dynamic pressure at V_p.

V_p = pull-up speed, feet per second.

(2) The applied unit loading in either direction need not exceed that corresponding to an applied control force of 200 pounds, applied as specified in section 31 (A), and it shall not be less than that corresponding to the minimum applied control force specified in sec. 31.

(*A*) (2) or an average value of 15 pounds per square foot, referred to the entire horizontal tail area.

(3) The chord distribution shall be in accordance with figure 5.

(*C*) *Supplementary condition.—*

(1) The total applied loads acting on the stabilizer portion, as determined from paragraph (B), shall also be applied in accordance with the load distribution illustrated in figure 4. In general, this loading will be less critical than that specified in paragraph (A).

Note. — No opposite load from the elevator is required in this case.

Sec. 27. Vertical Tail Surfaces

(*A*) *Maneuvering loads*.[1]—

[1]Where the engines are not in the plane of symmetry, the design speed for the vertical tail surfaces shall not be less than the maximum speed in level flight with one engine dead. See Aeronautics Bulletin No. 26, sec. 26 (F).

(1) Vertical tail surfaces shall be designed for an *applied* unit loading not less than that corresponding to the pull-up speed V_p (specified in sec. 26 (B) (1) (a)) and a normal force coefficient of 0.45.

(a) $$\bar{w} = 0.45\, q_p = \frac{V_p^2}{1870}.$$

Where = average applied unit load over entire vertical surface (other symbols as in sec. 26 (B) (1) (b)).

(2) For conventional control surfaces, the applied unit loading for this condition need not exceed that corresponding to an applied control force of 200 pounds, and it shall not be less than an average value of 12 pounds per square foot, referred to the entire vertical tail area.

(3) The chord distribution shall be in accordance with figure 5 and the load shall be assumed to act in either direction.

(*B*) *Supplementary conditions.—*

(1) The total applied load acting on the fixed surface (fin) as determined from paragraph (A) shall also be applied in accordance with the load distribution illustrated in figure 4, acting in either direction.

(2) The average applied load determined from paragraph (1) shall be not less than the value developed in encountering an ideal horizontal gust of 15 feet per second at the speed VL. (See sec. 14.) The following approximate formula may be used to determine the average applied unit loading for this condition:

(a) $$\bar{w} = \frac{V_L}{13}\left(\frac{4}{3+6/R}\right)$$

Where V_L is in feet per second.

R= aspect ratio of tail surface = $\frac{(\text{span})^2}{\text{area}}$

Note.— When *R* is less than 2.0, use 2.0.

(3) A special ruling shall be obtained when an automatic pilot is to be used on multiengined airplanes.

Sec. 28. Ailerons[2]

[2]Where the engines are not in the plane of symmetry, the design speed for the ailerons shall not be less than the maximum speed in level flight with one engine dead. (See Aeronautics Bulletin No. 26, sec. 26 (F).)

(*A*) Ailerons shall be designed for an *applied* unit loading not less than that corresponding to the pull-up speed Vp (specified in sec. 26 (B) (1) (a)) and a normal force coefficient of 0.45.

(1) $$(1)\quad \bar{w} = 0.45\, q_p = \frac{V_p^2}{1870}.$$

Where = average applied unit load over the aileron (other symbols as in sec. 26 (B) (1) (b)).

(*B*) For conventional ailerons, the applied unit loading for this condition need not exceed that corresponding to an applied control force couple of 80 pounds times the control wheel diameter, or a force of 80 pounds applied at the grip of the control stick, the control force to be resisted entirely by one aileron. The average applied unit loading shall not be less than 12 pounds per square foot.

(*C*) The chord distribution shall be in accordance with figure 6 and the load shall be assumed to act in either direction.

(*D*) When the aileron chord is relatively large, a check of the unit loading in the symmetrical accelerated flying conditions may be required, especially when such loads are equalized through an interconnecting tube which resists up loads in compression.

Sec. 29. Auxiliary Devices[1]

[1]See also Aeronautics Bulletin No. 26, sec. 27.

(*A*) *Wing flaps.* — The applied unit loading for wing flaps shall be determined from the most severe combination of airplane speed and flap normal force coefficient likely to be obtained during the operation or use of the flap. When a modified design speed for the extended flap condition is used in the wing analysis (as outlined in sec. 18 (D) (3)) that speed may be used also for the design of the flap. The values of flap force coefficients and the distribution of flap load shall be obtained from reliable test data. In general, a uniform load distribution over the flap is satisfactory for the design of the flap and its control system. The maximum normal force coefficient for the flap will not usually exceed 1.60, which corresponds to an applied unit loading of pounds per square foot, computed as follows:

(1) $$\bar{w} = 1.60\, q_f = \frac{V_f^2}{525}$$

Where q_f = dynamic pressure at V_f.

Vf = modified design speed for extended flaps, feet per second.

(*B*) *Trailing-edge tabs.* — The applied unit loading for control surface tabs shall be determined from the most severe combination of airplane speed and tab normal force coefficient likely to be obtained for any permissible loading condition of the airplane and at speeds up to the design gliding speed. In determining the design conditions for the tab and the entire control surface, consideration shall be given to the intended purpose of the tab (trimming, balancing, etc.), its relative size and range of movement, combined control system forces and tab action, and type of tab control system used (slow or rapid). In lieu of a complete investigation of possible tab design conditions, the following procedure may be employed in most cases.

(1) Determine the normal force coefficient for the tab fully deflected, assuming the deflection and angle of attack of the major control surfaces to be zero.

(2) Determine the applied unit loading, , on the tab, corresponding to the tab normal force coefficient and the high speed of the airplane in level flight:

(*a*) $$\bar{w} = C_n V_L^2/840.$$

Where V_L is in feet per second.

is in pounds per square foot.

(*b*) The load distribution may be assumed to be uniform over the tab.

(3) Determine the applied loading which, when distributed over the major control surface in accordance with figure 5, will balance the sum of the hinge moments produced by the tab load and the following control force applied by the pilot:

(*a*) For horizontal tail surfaces, the minimum value specified in section 31 (A) (2).

(*b*) For vertical tail surfaces on single-engined airplanes, zero.

(*c*) For vertical tail surfaces on multiengined airplanes (engines not in plane of symmetry), 200 pounds. (A lower load will be considered when tabs are relatively large and effective.)

(*d*) For ailerons, zero.

(4) For tail surfaces (not ailerons) determine the control surface loading corresponding to deflection of the movable surface and the tab in the same direction, as follows:

(*a*) For all portions of the movable surface except the area between the hinge line and the tab, assume the usual triangular load distribution (fig. 5) with an unknown value of unit loading, *w*, at the hinge line.

(*b*) For that portion of the movable surface included between its hinge line and the trailing edge of the tab, assume a uniform distribution. The unit pressure, w, at the hinge line of the movable surface should be assumed to be the same as in (a).

(*c*) Compute the moment about the main hinge line in terms of w and equate to the moment produced by a 200-pound load at the control column or rudder pedal to find the numerical value of w. This condition applies only to the movable surface and may be critical for the torque tube and members supporting the tab.

(*C*) *Special devices.*— Special rulings shall be obtained in connection with the design and analysis of wing-slot structures, spoilers, unconventional ailerons, auxiliary airfoils, and similar devices. Requests for special rulings should be accompanied by suitable drawings or sketches of the structure in question, together with general information and an outline of the method by which it is proposed to determine the design loads for the structure. In general, the procedure outlined in paragraphs (A) and (B) of this section can be modified to suit the particular case.

Sec. 30. Special Requirements

(*A*) *Cantilever surfaces.* — To obtain increased rigidity, internally braced tail surfaces shall have an additional (multiplying) factor of safety of at least 1.25 for all members which are critical with respect to the ultimate tensile stress or the modulus of rupture. This does not apply to other members such as those which fail through buckling or instability. In certain types of construction, even greater factors of safety may be necessary to secure sufficient rigidity.

(*B*) *Wire-braced surfaces.*—

(1) The sizes of and factors of safety for external brace wires shall be determined in accordance with section 62.

(2) All members affected by rigging loads shall be designed to provide the minimum factor of safety of 1.5 against the rigging loads computed in accordance with section 62.

(3) If the design is such that the counter wires do not become slack before the design load is reached, the superimposed loads from the counter wires should be accounted for in the analysis or suitable conservative assumptions should be made.

(*C*) *Tests.*—

(1) In addition to the analyses specified in sections 26 to 29, the strength and rigidity of control surfaces and auxiliary devices shall be demonstrated by proof tests (as defined in sec. 11) in the presence of a Department of Commerce representative. Control surface tests shall include the horn or fitting to which the control system is attached.

(2) *Strength tests* as defined in section 11 will be accepted in lieu of stress analyses for control surfaces and auxiliary devices, provided that such tests are carried to destruction or 125 percent of the design load. The design load for such tests may be based on the minimum factor of safety used, provided that a partial stress analysis or additional supplementary tests are conducted to substantiate the ultimate strength of members for which additional factors of safety are required. Structural components which have undergone strength tests shall not be incorporated in the airplane structure unless satisfactory evidence is submitted to prove that the structure was not damaged in the tests.

(3) For the tests described in paragraphs (1) and (2) all external brace wires shall be rigged to approximately the same tension as would be employed in normal use.

(*D*) *Chord load.*— Control surfaces shall be designed to withstand a reasonable amount of chord load in either direction. A total chord load equal to 20 percent of the maximum normal load may be used as a separate design condition. The distribution along the span may be made proportional to the chord, if desired. Tests for this condition are not required unless the structure is such as to indicate the advisability of such tests.

(*E*) *Installation.* —

(1) Movable tail surfaces shall be installed so that there is no interference between the surfaces or their bracing when one is held in its extreme position and the other operated through its full angular movement.

(2) When an adjustable stabilizer is used, stops shall be provided at the stabilizer to limit its movement to a range equal to or slightly greater than the maximum required to balance the airplane.

(*F*) *Hinges.* — Hinges of the strap type bearing directly on torque tubes are permissible only in the case of steel torque tubes which have an indicated margin of safety of at least 50 percent at the hinge point. In other cases sleeves of suitable material shall be provided for bearing surfaces. Control surface hinges (excepting ball or roller bearings) shall incorporate a factor of safety of at least 10 with respect to the ultimate bearing strength of the softest material used as a bearing. The manufacturer's rating (non-Brinnell) for ball or roller bearings shall equal or exceed the *applied* load on the bearing. Clevis pins may be used as hinge pins provided they are made of material conforming with, or the equivalent of, S.A.E. Specification 2330.

(*G*) *Flutter prevention.*[1] — The general principles of flutter prevention should be observed on all airplanes. This applies particularly to the design and installation of control surfaces and control systems and includes such desirable features as structural stiffness, elimination of all play in hinges and control system joints, rigid interconnections between ailerons and between elevators, a relatively high degree of weight balance of control surfaces, a relatively low amount of aerodynamic balance, high frictional damping, and adequate wing fillets and fairing. Features tending to create aerodynamic disturbances, such as sharp leading edges on movable surfaces, should be carefully avoided. These principles apply also to wing flaps and particularly to control surface tabs, which are relatively powerful and correspondingly more dangerous if not properly designed. It should be realized that various forms of flutter are possible and that there usually exists for each type of flutter a critical speed at which it will begin. This critical speed will be raised by any improvement in the anti-flutter characteristics of the particular portion of the airplane involved and may even be eliminated entirely in some cases. Not all of the previously named aids to flutter prevention are necessary in combination, as the desired result can often be achieved by utilizing only certain features to a sufficient degree. The following requirements, however, represent the flutter-prevention measures which are considered essential and shall be complied with unless special means for preventing flutter are provided which are satisfactory to the Department.

[1] See also Aeronautics Bulletin No. 26, sec. 29.

For all airplanes —

(1) When separate elevators are used they shall be interconnected. The connecting structure should be as stiff in torsion as practicable.

(2) Ailerons attached to internally braced wings or to wings braced by wires only shall be statically balanced about their hinge lines.

Note. — If a balancing weight is employed, its effectiveness is usually increased by locating it near the outboard end of the aileron. Partial static balance may be satisfactory in cases where the aileron control system is irreversible and exceptionally rigid.

(3) Trailing-edge tab controls shall be irreversible, relatively rigid, and the tab installation shall be such as to prevent any development of free motion of the tab. Irreversible controls are not required when tabs are completely statically balanced, provided that the tab control system incorporates some means for preventing too abrupt

manipulation of the tab through a large range. A small amount of static *overbalance* of the tab (C.G. ahead of hinge) will tend to offset any possible harmful effects of tab looseness or tab control system flexibility.

(4) Where trailing-edge tabs are used to assist in moving the main surface (balancing tabs), care shall be taken in proportioning areas and relative movements so that the main surface is not aerodynamically overbalanced at any time.

(5) Experimental determination of the natural frequency of vibration of certain components of the airplane may be required in cases where there are indications of dangerously low frequencies or of coincidence of the natural frequencies of two or more separate structural components.

For airplanes having a high speed greater than 150 miles per hour —

(6) The dynamic balance coefficient of the rudder and of each separate elevator as computed about the point of intersection of the hinge line with the center line of the fuselage shall be not more than 0.08. This coefficient is nondimensional and consists of a fraction whose numerator is the resultant product of inertia of the control surface and whose denominator is equal to the mass (or weight) multiplied by the aerodynamic area of the control surface. The resultant product of inertia is obtained from the algebraic sum of the products of inertia in the four quadrants, considering the products of inertia in the first and third quadrants as positive and that in the second and fourth quadrants as negative. A dynamically unbalanced surface will always have a positive resultant product of inertia. If it can conveniently be done, the rudder should be completely dynamically balanced. The computation of the dynamic balance coefficient shall be submitted with the stress analysis of the pertinent control surface.

Chapter V. CONTROL SYSTEMS[1]

[1] See Aeronautics Bulletin No. 26, ch. V.

Sec. 31. Loading Conditions

(*A*) For *elevator control systems,* the applied loading shall be 25 percent greater than that corresponding to the specified applied loading for the elevator, assuming the elevator to be in that deflected position which requires the greatest load on the control column for balance, with the following exceptions.

(1) The applied loading *need not exceed* that corresponding to a maximum force of 200 pounds applied at the grip of a control stick, or forces of half this value applied at two diametrically opposite points on the rim of a control wheel and acting in a fore-and-aft direction.

(2) The applied loading *shall not be less than* that corresponding to a minimum force applied as in paragraph (1) and determined as follows:

$F = 70+0.06\ (W-500.)$

Note. — F need not be greater than 130 pounds.

W = airplane gross weight in pounds

(*B*) For *rudder control systems,* the applied loading shall be 25 percent greater than that corresponding to the specified applied loading for the rudder, assuming the rudder to be in the fully deflected position, with the following exceptions:

(1) The applied loading *need not exceed* that corresponding to a maximum force of 200 pounds applied at the point of contact of the pilot’s foot.

(2) The applied loading *shall not be less than* that corresponding to a minimum force of 130 pounds applied as in paragraph (1).

(3) For multi-engined airplanes (engines not in the plane of symmetry) the applied loading shall correspond to a control force of 200 pounds, applied as in paragraph (1).

(*C*) For *aileron control systems,* the applied loading shall be 25 percent greater than that corresponding to the specified applied loading for the ailerons, assuming the ailerons to be simultaneously loaded in opposite directions and fully deflected. The following exceptions and rules apply in this case.

(1) The applied loading *need not exceed* that corresponding to a maximum force of 80 pounds applied at the grip of a control stick, or a couple equal to the product of the diameter of the control wheel and a force of 80 pounds.

(2) The applied loading *shall not be less than* that corresponding to a minimum force, applied as in paragraph (1) and determined as follows:

$F = 30 + 0.02\ (W\text{-}500.)$

Note. — F need not be greater than 50 pounds.

W = airplane gross weight in pounds.

(3) The following separate assumptions shall be made when conventional (nondifferential) ailerons are used:

(*a*) Seventy-five percent of the stick force or couple shall be assumed to be resisted by a "down" aileron, the remainder by the other aileron.

(*b*) Fifty percent of the stick force or couple shall be assumed to be resisted by an "up" aileron, the remainder by the other aileron.

(4) When differential ailerons are used it shall be assumed that 75 percent of the stick force or couple is resisted by each aileron in either the "up" or "down" position.

(*D*) Stress analyses shall be submitted for control systems used to operate flaps, tabs, and other similar devices. The control system applied loads shall be at least 25 percent greater than the applied loads from the control surfaces. The general principles previously outlined can be applied, with suitable modifications. Preliminary approval of the proposed method of analysis should be obtained in doubtful cases.

(*E*) The forces in the control wires or push rods operating the movable surfaces shall be computed and their effect on the rest of the structure shall be investigated and allowed for in the design of the structure.

Sec. 32. Bungee Devices

(*A*) The use of bungee devices will be permitted within the following limitations:

(1) In all cases the airplane shall be satisfactorily maneuverable and controllable under all conditions without assistance from the bungee device.

(2) In all cases the bungee mechanism shall be of a type and design satisfactory to the Secretary.

(3) Rubber cord shall not be used for this purpose.

Sec. 33. Special Requirements

(*A*) *Tests.* — In addition to the required stress analyses, the strength and rigidity of control systems shall be demonstrated by *proof tests* (as defined in sec. 11) in the presence of a Department of Commerce representative. When a control system is tested in conjunction with the control surface which it operates, any difference between the required loadings shall be accounted for by the application of the necessary force or couple at the control horn, or its equivalent. The direction of the test load shall be such as to produce the most critical loadings for the control system members involved. The test shall include all fittings, pulleys, and brackets used to attach the control system to the primary structure.

(*B*) An *operation test* shall be conducted by operating the controls from the cockpit with 50 percent of the applied load on the surfaces, except that the load so applied shall not be less than that corresponding to the minimum control force specified for the design of the control system (see sec. 31, pars. (A) (2), (B) (2) and (C) (2)). The control surface load shall be distributed so as to approximate the required load distribution. In this test there shall be no jamming, excessive friction, or appreciable deflection.

(*C*) *Strength tests* as defined in section 11 will be accepted in lieu of stress analysis for control systems, provided that all essential parts of the control systems are included in the test and that the strength of the supporting airplane structure is also substantiated either by test or by stress analysis.

(*D*) *Stops*. — All control systems shall be provided with stops which positively limit the range of motion of the control surfaces. Stops shall be capable of withstanding the loads corresponding to the design conditions for the control system.

(*E*) *Installation.*— Proper precautions shall be taken to eliminate the possibility of jamming of the control system, interference from cargo or passengers, and chafing or slapping of cables against parts of the airplane. All pulleys shall be provided with satisfactory guards.

(*F*) *Welds.* — In control systems welds shall not be employed to carry tension without reinforcement from rivets or bolts. Direct welding of control horns to torque tubes (without the use of a sleeve) is permissible only when a large excess of strength is indicated.

(*G*) *Joints.*— Bolts with castellated nuts safetied with cotter pins or an approved type of self-locking nut shall be used throughout the control system. Pins and bolts which are used in control systems shall be of uniform material, of high quality, and of first-class workmanship.

Note. — The use of clevis pins in standard cable ends is considered satisfactory for light airplanes.

(*H*) All joints subjected to angular motion (excepting control surface hinges and joints incorporating ball or roller bearings) shall incorporate a factor of safety of at least 5 with respect to the ultimate bearing strength of the softest material used as a bearing. The manufacturer's rating (non-Brinell) for ball or roller bearing shall equal or exceed the *applied* load on the bearing.

Chapter IV. LAND-TYPE LANDING GEAR

Sec. 34. Design Conditions

(*A*) The following conditions represent the minimum amount of investigation required for conventional landing gear. For unconventional types it may be necessary to investigate other landing attitudes, depending on the arrangement and design of the landing gear members. Consideration will be given to a reduction of the specified load factors in cases where it can be proved that the shock-absorbing system will positively limit the acceleration factor to a definite lower value.

Sec. 35. Level Landing

(A) The minimum level landing applied load factor shall be obtained as follows:

(1) *For airplanes of over 1,000 pounds gross weight.* —

$$n = 2.80 + \frac{9{,}000}{W + 4{,}000}.$$

Where W = gross weight, in pounds.

(2) *For airplanes of less than 1,000 pounds gross weight.* —

$n = 3.00 + 0.133\,s$.

Where s = wing loading, in pounds per square foot.

(3) In any case the value of applied load factor n as determined from (1) or (2) need not exceed 4.33.

(B) The resultant of the ground reaction shall be assumed to be a force lying at the intersection of the plane of symmetry and a plane in which are located the axles and the center of gravity of the airplane less chassis. The propeller axis (or equivalent reference line) shall be assumed horizontal and the basic value of the vertical component of the resultant of the ground reaction shall be equal to the gross weight of the airplane minus chassis and wheels. The horizontal component shall be of the magnitude required to give the resultant force the specified direction, except that it need not be greater than 25 percent of the vertical component. When this exception applies, the resultant force will not pass through the center of gravity. Methods of accounting for the unbalanced moment are outlined in Aeronautics Bulletin No. 26, section 39 (C).

(C) If a sliding element instead of a rolling element is used for the landing gear, a horizontal component of one-half of the vertical component shall be used to represent the effect of ground friction, except that ski gear which are designed and used only for landing on snow and ice may be designed for the same horizontal component as wheel gear.

(D) The resultant of the ground reaction shall be assumed to be divided equally between wheels and to be applied at the axle at the center of the wheel. The shock-absorber unit and tires shall be assumed to be deflected to half their total travel, unless it is apparent that a more critical arrangement could exist.

Sec. 36. Three-Point Landing

(A) The applied load factor shall be the same as for the level landing condition.

(B) The basic value of the sum of the ground reactions shall be the gross weight of the airplane less chassis. The total load shall be divided between the chassis and tail skid in inverse proportion to the distances, measured parallel to the ground line, from the center of gravity of the airplane less chassis to the points of contact with the ground. The load on the chassis shall be divided equally between wheels. Loads shall be assumed to be perpendicular to the ground line in the 3-point landing attitude, with all shock absorbers and tires deflected to the same degree as in the level landing condition. The tail wheel or skid installation shall also be investigated for this condition.

Sec. 37. Side Load and One-Wheel Landing Conditions

(A) *Side load.* — The landing gear and adjacent portions of the fuselage and cabane shall be analyzed for the following loading:

(1) An *applied* load equal to the gross weight of the airplane (less chassis) times one-tenth of the level landing applied load factor shall be assumed to act inward at the point of contact of one wheel with the ground in a direction normal to the plane of symmetry.

(2) The load specified in paragraph (1) shall be balanced by inertia loads, the resultant of which passes through the center of gravity of the airplane less chassis. These loads can be distributed between panel points adjacent to the landing gear.

(3) The unbalanced moment of the side load about the center of gravity will be resisted by angular inertia loads. It will be satisfactory to apply the necessary couples at the wing attachment points, assuming that the unbalanced turning moment is resisted entirely by the angular inertia of the wing structure.

(4) The shock absorbers and tires shall be assumed to be deflected to the same degree as in level landing.

(5) No vertical or drag loads need be assumed in conjunction with the side load.

(*B*) *One-wheel landing.*— An investigation of the fuselage structure is required for a one-wheel landing, in which only those loads obtained on one side of the fuselage in the level landing condition are applied. The resulting load factor is therefore one-half of the level landing load factor. This condition is identical with the level landing condition insofar as the landing gear structure is concerned. The unbalanced rolling moment will be resisted by angular inertia loads which are supplied largely by the wing structure. See Aeronautics Bulletin No. 26, section 34, for additional information on this subject.

Sec. 38. Braked Landing

(*A*) The applied load factor for this condition shall be 1.33.

(*B*) Airplanes equipped with brakes shall be investigated for the loads incurred when a landing is made with the wheels locked and the airplane is in an attitude such that the tail skid just clears the ground. The weight of the airplane minus the weight of the chassis shall be assumed to act on the wheels in a direction perpendicular to the ground line in this attitude. In addition, a component parallel to the ground line shall be assumed to act as the point of contact of the wheels and the ground, the magnitude of this component being equal to the weight of the airplane, minus chassis, times a coefficient of friction of 0.55. The tire in all cases shall be assumed to have deflected not more than one-quarter the nominal diameter of its cross section, and the deflection of the shock absorbers shall be the same as in level landing.

Sec. 39. Shock Absorption Requirements[1]

[1] See also Aeronautics Bulletin No. 26, sec. 35.

(*A*) The height of free drop, in inches, shall be 0.36 times the calculated stalling speed in miles per hour, except that it need not be greater than 18 inches for conventional airplanes. When auxiliary high-lift devices or “air brakes” are employed, the height of free drop shall not be less than 18 inches. (A special investigation of the energy to be absorbed may be required in the case of airplanes having exceptionally high-sinking speeds, pending a general revision of shock absorption requirements to include basic airplane characteristics.) The height of free drop is measured from the bottom of the tires to the ground, with the landing gear extended to its extreme unloaded position. If the design of the shock-absorbing system is such that the above method of specifying the required energy absorption capacity appears to give irrational results, an alternate method will be considered upon presentation of the pertinent data.

(*B*) The shock-absorbing systems, including that for the tail, shall be capable of absorbing the kinetic energy generated during the entire vertical travel of the airplane when the airplane is dropped from the required height of free drop, without subjecting any structural member to a load greater than its maximum design load. This applies to both the level landing and three-point landing conditions. The effects of support from the wings and the shock-absorbing capacity of the structure shall be neglected in determining the energy to be absorbed by the shock-absorbing system.

(*C*) The shock-absorbing units used in the shock-absorbing systems must conform to the requirements for approval outlined in Aeronautics Bulletin No. 7-F except that a drop test of the complete fuselage and landing gear assembly, under full-load conditions and with a suitable accelerometer or other satisfactory means of recording impact forces installed, will be acceptable as evidence of the shock-absorbing capacity of the strut.

Sec. 40. Wheels

(*A*) Landing gear wheels shall be of a type and design approved by the Secretary in accordance with the requirements outlined in Aeronautics Bulletin No. 7-F and shall not be subjected to loading conditions in excess of those for which they are approved. The static working load of a landing gear wheel shall be taken as half the gross weight of the airplane upon which it is installed. The static working load for a tail wheel shall be taken as the maximum actual load which may be imposed on the tail wheel when the airplane is at rest on the ground.

(*B*) When the wheels installed on a specific airplane are of such a size as to require that a tire of special construction be used to support the airplane weight, *the wheels shall be plainly and conspicuously marked to that effect.* Such marking shall state the make and size of the proper tire and the number of plies required in its construction.

Sec. 41. Special Considerations.

(*A*) The structural members of wire-braced landing gear shall be designed to provide a factor of safety of at least 1.5 against the rigging loads computed in accordance with section 62 (A), (B), and (C). For landing gear structures which are a part of a wing lift truss, section 62 (D) shall be complied with.

(*B*) The tail skid or tail wheel installation may be dropped from the height specified in section 39 in lieu of submitting a tail skid or wheel installation analysis, if the manufacturer so desires.

(*C*) When retractable landing wheels are used, means shall be provided for indicating to the pilot, at all times, the position of the wheels. Separate indicators for each side are required unless a single indicator is obviously satisfactory. A positive lock shall be provided for the wheels in the extended position, unless a rugged irreversible operating mechanism is used.

(*D*) Care shall be taken to avoid stress concentrations in members subjected to dynamic loads, especially those which have been highly heat-treated. All fillets in members subjected to bending shall be made as large as is practicable.

(*E*) Landing gear members which also form part of a lift truss shall be designed in accordance with section 25 (*F*).

Chapter VII. BOAT HULLS AND SEAPLANE FLOATS[1]

[1] See Aeronautics Bulletin No. 26, ch. VII.

Sec. 42. Float Requirements

Seaplanes shall be equipped with floats of a type and design approved by the Secretary in accordance with Aeronautics Bulletin No. 7-F, in which are outlined the design requirements for seaplane floats.

Sec. 43. General Design Conditions

(*A*) *For float-type seaplanes,* the float supporting structure shall be investigated for the landing conditions outlined in sections 44, 45, and 46.

(*B*) *Flying-boat hulls* shall be investigated for the loads imposed by the wings and tail surfaces (including maneuvering loads) in the various flight conditions specified elsewhere in these regulations. The landing conditions specified in section 47 and the bottom loading conditions specified in section 48 shall also be investigated.

Note. — Consideration will be given to alternate methods of determining the external applied loading on flying boat hulls, pending a general revision of the requirements for hulls and floats.

(*C*) The landing conditions outlined in section 44, 45, and 46 apply also to fuselages of float-type seaplanes. Wing structures supporting considerable dead weight may also be critically loaded in the landing conditions.

Sec. 44. Landing With Inclined Reactions (Seaplanes)

(*A*) The applied load factor for seaplanes of 1,000 pounds gross weight or more shall be 4.20 vertically.

(*B*) For seaplanes weighing less than 1,000 pounds gross weight the applied load factor shall be the same as that required for a land plane of the same gross weight and wing loading.

(*C*) The propeller axis (or equivalent reference line) shall be assumed to be horizontal and the resultant water reaction acting in the plane of symmetry and passing through the center of gravity of the airplane less floats and float bracing, but inclined so that its horizontal component is equal to one-quarter of its vertical component. The forces representing the weights of and in the airplane act in a direction parallel to the water reaction. The weight of the floats and float bracing may be deducted from the gross weight of the airplane.

(*D*) For the design of float attachment members, including the members necessary to complete a rigid brace truss through the fuselage, the minimum factor of safety shall be 1.85.

(*E*) For the remaining structural members the minimum factor of safety shall be 1.50.

Sec. 45. Landing With Vertical Reactions (Seaplanes)

(*A*) The applied load factor shall be 4.33 vertically for seaplanes having a gross weight of 1,000 pounds or more. For seaplanes weighing less than 1,000 pounds, the applied load factor shall be the same as that required for a land plane of the same gross weight and wing loading.

(*B*) The propeller axis (or equivalent reference line) shall be assumed to be horizontal, and the resultant water reaction vertical and passing through the center of gravity of the airplane less floats and float bracing. The weight of the floats and float bracing may be deducted from the gross weight of the airplane.

(*C*) The minimum factors of safety shall be the same as those specified in section 44 for landing with inclined reactions.

Sec. 46. Landing With Side Load (Seaplanes)

(*A*) An applied load factor of 4.0 vertically is required, to be applied to the gross weight of the airplane less weight of floats and float bracing.

(*B*) The propeller axis (or equivalent reference line) shall be assumed to be horizontal and the resultant water reaction shall be assumed to be in the vertical plane which passes through the center of gravity of the airplane less floats and float bracing and is perpendicular to the propeller axis. The vertical load shall be applied through the keel or keels of the float or floats, evenly divided between the floats when twin floats are used. A side load equal to one-fourth of the vertical load shall be applied along a line approximately half way between the bottom of the keel and the level of the water line at rest. When built-in struts are used, check calculations shall be made for the built-in struts with the side load at the level of the water line at rest. When two main floats are used, the entire side load specified shall be applied to the float on the side from which the water reaction comes.

(*C*) The minimum factor of safety shall be 1.50.

Sec. 47. Boat Hull Landing Loads (Flying boats)

The hull structure, acting as a unit, shall be designed for the external applied loads determined by the following separate conditions.

(*A*) *Step landing.* — The aircraft shall be in such an attitude that the propeller axis (or equivalent reference line) is horizontal and shall be assumed to be supported by a vertical buoyant force distributed over an area extending from the step forward to a point half way between the step and the forward end of the normal load water line. This area may be assumed to be a rectangle whose width is equal to the full projected width of the bottom of the step. The load on the area shall be so distributed that its intensity is 50 percent greater at the keel than at the chine and 50 percent greater on the section at the step than at the forward section. The volume of the prismoidal loading curve so obtained, from which the intensities may be computed, shall equal the gross weight of the airplane times an applied load factor of 5.33.

(1) For this condition and load factor.

$$nW = \frac{25abL}{16}.$$

Where n = applied load factor = 5.33.

W = gross weight of airplane.

a = intensity of loading at the chine at the forward section.

b = the beam.

L = half the length of the L.W.L forward of the step.
The centroid of this loading may be assumed to be on the keel from the forward edge of the load, and the resultant water reaction shall pass through this centroid and the center of gravity of the airplane.

(2) The forces representing the weights of and in the airplane shall act in a direction parallel to the water reaction.

(3) The minimum factor of safety shall be 1.50.

(*B*) *Two-wave landing.* — The aircraft shall be assumed to be supported by vertically applied up loads at each end of the L.W.L., the magnitude of each load being such that the resultant *applied* load passes through the main step and equals the gross weight of the airplane. The applied shears and bending moments shall be computed by assuming the gross weight of the airplane to be concentrated at the step and omitting all panel-point loads.

(1) The minimum factor of safety shall be 1.50.

(2) The structure at the point of application of the external loads need not be investigated for local stresses, in this condition.

Sec. 48. Bottom Loading (Boat Hulls)

(*A*) The bottom plating stringers, frames, and adjoining structure shall be investigated for the *applied* unit loading determined by the following conditions, using the minimum factor of safety of 1.50.

(1) The applied unit loading for that portion of the hull bottom just forward of the main step shall be that determined from section 47 (A).

(2) The area from the forward end of the normal load water line to a point half-way between the step and the forward end of the normal load water line shall be designed to support an applied load having the unit pressure found at the forward portion of the chine in section 47 (A).

(3) The area extending from the step to the rear end of the normal load water line shall be designed to support an applied load having a unit pressure equal to 75 percent of the average *applied* unit pressure found in section 47 (A).

Sec. 49. Buoyancy

(*A*) The hulls of flying boats and amphibians shall be divided into water-tight compartments in accordance with the following:

(1) In aircraft of 5,000 pounds gross weight or more the compartments shall be so arranged that, with any two adjacent compartments flooded, the hull and wing tip floats (and tires, if used) will retain sufficient buoyancy to support the gross weight of the aircraft in fresh water.

(2) In aircraft of 1,500 to 5,000 pounds gross weight the compartments shall be so arranged that, with any one compartment flooded, the hull and wing tip floats (and tires, if used) will retain sufficient buoyancy to support the gross weight of the aircraft in fresh water.

(3) For aircraft of less than 1,500 pounds gross weight, water-tight subdivision of the hull is not required.

(4) Bulkheads may have water-tight doors for the purpose of communication between compartments.

(*B*) Wing-tip floats shall be so arranged that, when completely submerged in fresh water, they will provide a righting moment which is at least 1.5 times the upsetting moment caused by the aircraft being tilted. A greater degree of stability may be required in the case of large flying boats, depending on the height of the center of gravity above the water level, area and location of wings and tail surfaces, and other considerations.

Sec. 50. Wing-Tip Floats

(*A*) Wing-tip floats and their attachment, including the wing structure, shall be analyzed for each of the following conditions:

(1) An applied load acting vertically up at the completely submerged center of buoyancy and equal to three times the completely submerged displacement.

(2) An applied load inclined upward at 45° to the rear and acting through the completely submerged center of buoyancy and equal to three times the completely submerged displacement.

(3) An applied load acting parallel to the water surface (laterally) applied at the center of area of the side view and equal to one and one-half times the completely submerged displacement.

(*B*) Wing-tip float attachment members shall be designed so that breaking loads will cause failure of the attachment members before damage to the primary wing structure occurs.

Chapter VIII. FUSELAGE, ENGINE, MOUNTS, AND NACELLES[1]

[1] See Aeronautics Bulletin No. 26, ch. VIII.

Sec. 51. Fuselage Design Conditions

(*A*) The fuselages of all airplanes shall be investigated for the loads imposed by the wings, landing gear, tail surfaces (including maneuvering loads), and engine in the various flight and landing conditions specified elsewhere in these regulations. The balancing tail loads determined from the accelerated flight conditions shall be used in the fuselage analysis. Any condition which can be shown to be noncritical for the fuselage need not be investigated. In some cases only a certain portion of the fuselage structure need be investigated.

(*B*) The additional fuselage design conditions outlined in the succeeding sections shall also be investigated.

(*C*) Unless otherwise specified, a minimum factor of safety of 1.5 shall be maintained.

Sec. 52. Nosing Over

(*A*) The *applied* load factor required shall be that for the 3-point landing condition.

(*B*) The forward portion of the fuselage shall be investigated for this condition. The following approximate method may be used, unless the design is such that a more rational method appears to be desirable. Assume the airplane to be resting on the wheels and that portion of the primary structure of the fuselage or engine which would strike the ground first. Assume the gross weight of the airplane to act at the center of gravity and perpendicular to the ground. The forward portion of the fuselage should then be designed to withstand the ground reaction thus obtained.

Sec. 53. Complete Turn-Over

(*A*) The minimum *applied* load factor for this condition shall be 3.0.

(*B*) The fuselage and cabane structure shall be designed to protect the pilot and passengers in case the airplane should turn completely over onto its back. For this purpose the cabane, or equivalent structure, shall be designed to carry an applied load acting normal to the thrust line (or equivalent reference line) and equal to three times the gross weight of the airplane. In cases where a wing is above the fuselage and braced by more than 1 cabane lift truss, at least 1 truss shall be designed for the entire applied load. The superstructure shall also be capable of resisting a total applied load of at least one-half the weight of the airplane, acting either forwardly or rearwardly parallel to the thrust line or wing chord and suitably divided between the uppermost points of the side trusses of the cabane or equivalent structure. In other types of design a structure of equivalent strength shall be provided that will prevent the heads of the pilot and passengers from striking the ground when the airplane is resting on this structure and the top of the fin post. In all cases provision shall be made to permit egress of the pilot and passengers in the event of this type of accident.

Note — Special consideration will be given to types of aircraft for which the possibility of a complete turn-over in landing is remote.

Sec. 54. Torque

(*A*) In the case of direct-drive engines having five or more cylinders the stresses due to the torque load shall be multiplied by an *applied* load factor of 1.5. For 4-, 3-, and 2-cylinder engines the applied load factors shall be 2, 3, and 4, respectively. The basic torque acting on the airplane structure shall be computed for the rated power of the engine and the propeller speed corresponding to the rated speed of the engine.

(*B*) The engine mount and forward portion of the fuselage and/or nacelles shall be designed for this condition.

Sec. 55. High Angle of Attack and Torque

(*A*) The torque loads determined from section 54 shall be considered as acting simultaneously with the loads determined for accelerated flight condition I. The engine mount and forward portions of the fuselage and/or nacelles shall be designed for this condition. A check shall be made for the gust load factor developed in flying with minimum load. (See sec. 15 B (1) a.)

Sec. 56. Side Load on Engine Mount

(*A*) The applied load factor for this condition shall be equal to one-third of the applied load factor for flight condition I but shall in no case be less than 1.33.

(*B*) The engine mount and forward section of the fuselage and/or nacelles shall be analyzed for this condition, considering the applied load to be produced by inertia forces.

Sec. 57. Up load on Engine Mount

For engine mounts to which the nosing-over condition does not apply, the following arbitrary condition shall be investigated:

(*A*) The design load in each member shall be assumed as 50 percent of that in the level landing condition but of opposite sign.

Sec. 58. Nacelle Design Conditions

(*A*) Nacelles and their supporting structure shall be analyzed for those flying and landing conditions which impose critical loads, including those conditions specified in this chapter which can be applied to nacelles.

(*B*) The required applied load factors and factors of safety are the same as those specified elsewhere for the specific loading conditions.

(*C*) A special investigation for the gust load factors obtained with minimum airplane gross weight should be made. (See sec. 55 (B).)

Sec. 59. Requirements for New Engine Mount Designs

(*A*) When a new engine mount is to be designed for an existing airplane and it is not necessary, because of the margins of safety in the original structure, to make an entire new analysis of the airplane, the following conditions shall be investigated:

(1) High angle of attack and torque, as specified in section 55.

(2) Nosing over or up load, as specified in sections 52 or 57.

(3) Side load, as specified in section 56.

(*B*) The analysis for each condition shall be complete enough to show that no negative margins of safety exist in the airplane structure.

Sec. 60. Special Requirements

(*A*) *Seats and safety belts.*— All seats shall be provided with approved safety belts. Safety belts and their attachments shall be capable of withstanding a design load of 1,000 pounds per person applied upwardly and forwardly at an angle of approximately 45° with the floor line. This load shall be carried through to the main structure. Seats or chairs, even though adjustable, in open or closed airplanes, shall be securely fastened in place whether or not the safety belt load is transmitted through the seat.

(*B*) *Stress analysis.* —

(1) Redundant structures shall be investigated by an exact method or by making conservative "overlapping" assumptions as to load distribution.

(2) The effects of local loads (such as are caused by control systems and dead weight) shall be investigated.

(3) A special investigation shall be made for all discontinuities in the fuselage trussing and for cut-outs in monocoque fuselages.

(4) The coefficient of restraint used in determining critical column loads shall in no case exceed 2.0. A value of 1.0 shall be used for all members in the engine mount. In doubtful cases, tests will be required to substantiate the degree of restraint assumed.

(5) The physical properties of welded metal members shall be reduced in conformity with accepted Army practice. (See also Aeronautics Bulletin No. 26, sec. 43.)

Chapter IX. MISCELLANEOUS STRUCTURAL REQUIREMENTS[1]

[1] See Aeronautics Bulletin No. 26, ch. IX.

Sec. 61. Requirements for Fittings and Standard Parts

(*A*) The term "fittings" is used to define all parts used to connect one primary member to another and includes the bearing of those parts on the members thus connected.

(*B*) *The minimum factor of safety for fittings shall be 1.80.* The *design* load for fittings therefore shall not be less than 1.80 times the *applied* load. The factor of safety for a given critical loading condition shall in any case be 20 percent greater than the specified minimum factor of safety for the primary member from which the fitting receives its critical load. The additional factors of safety required for certain primary members to provide increased rigidity need not be incorporated in fittings. (See also par. (H) of this section.) Direct welding between primary structural members does not constitute a fitting. In a fitting made up of several parts welded together, proper allowance for the effects of welding shall be made.

(*C*) Since the system of forces which designs a fitting does not necessarily include the forces which design the attaching members, all the forces acting in all the specified conditions shall be considered for every fitting. The strength of each part of a built-up fitting shall be investigated and proper allowance shall be made for the effects of eccentric loading, when present. A strength test may be required in cases where stress analysis methods appear inadequate.

(*D*) *Bolts and screws.*— All bolts and screws in the primary structure shall be of uniform material of high quality and of first-class workmanship. *Commercial machine screws* shall not be used as substitutes for aircraft bolts in any part of the primary structure. The use of an *approved locking device or method* is required for all bolts, screws, and pins.

(*E*) The allowable bearing load assumed for the threaded portion of a bolt shall not exceed 25 percent of the rated shear strength of the bolt. *Eyebolts* subject to bending shall have a radius of at least one-fourth of the diameter of the shank between the head and shank.

(*F*) The *use of wood screws* is prohibited for the attachment of such parts as control surface hinges, or fittings transmitting structural loads.

(*G*) *Tierods and wires. —* The minimum size of tierod which may be used in the primary structure is no. 6-40. The corresponding minimum allowable size of single-strand hard wire is no. 13 (0.072-inch diameter). The assumed terminal efficiency of single-strand hard wire shall not be greater than 85 percent.

(*H*) Fittings attached to wires and cables up to and including the 3,400-pound size shall have at least the same rated strength as the wires or cables. In the case of fittings to which several tierods or wires are attached, this requirement applies separately to each portion of the fitting to which a tierod or wire is attached, but does not require simultaneous application of rated wire loads.

(*I*) *Fabrication.— Brazing* is not permitted on assemblies constituting parts of the primary structure, except in certain cases. Typical preliminary drawings, samples, or test data should be submitted when approval of brazing is desired. The same requirements apply to *spot welding.*

(*J*) All fittings shall have adequate protective covering.

Sec. 62. Wire-Braced Structures[1]

[1] See Aeronautics Bulletin No. 26, sec. 42.

(*A*) The rigging load in a wire shall be expressed in percentage of the rated strength of the wire proper (not the terminal).

(*B*) In a wire-braced structure the wire sizes shall be such that the ratio between rigging loads in any two wires, expressed as in paragraph (A), shall not exceed 2.0.

(*C*) The strength of all members affected by the rigging loads in brace wires shall be sufficient to provide the minimum factor of safety of 1.5 with respect to the *applied* loads imposed by the wires when rigged so that the average of the rigging loads as defined in (A) is equal to 20 percent.

(*D*) For externally braced lifting surfaces or control surfaces the minimum factor of safety for the wire bracing shall be multiplied by an additional factor of safety to prevent excessive deflection, computed as follows:

$K = \frac{L}{2R}$ (shall not be less than 1.0).

Where K = the additional factor of safety.

R = the reaction representing the applied air load resisted by the wire.

L = The load required in the wire to furnish the reaction R.

(*E*) The requirement outlined in paragraph (D) will be waived in case the Army method for determining wire sizes is employed.

Sec. 63. Fabrication Methods

(*A*) All exposed wood structural members shall be given at least two protective coatings of varnish or approved equivalent. Built-up box spars and similar structures shall be protected on the interior by at least one coat of varnish or approved equivalent. Due care shall be taken to prevent coating the gluing surfaces.

(*B*) When casein glue is used, laminated spruce spar flanges, webs, or similar structures shall have a pressure of 100 or 150 pounds per square inch applied to the joint during the gluing process. In gluing hard woods for structural purposes a pressure of 200 to 250 pounds per square inch shall be used.

Chapter X. NONSTRUCTURAL REQUIREMENTS

Sec. 64. General.

(*A*) The cockpit shall be constructed to afford suitable ventilation and adequate vision of the pilot under normal flying conditions. In cabin aircraft the windows shall be so arranged that they may be cleaned or opened in flight to provide forward vision for the pilot. See Aeronautics Bulletin No. 7-J for special requirements for transport aircraft.

(*B*) The pilot and the primary control units, excluding cables and control rods, shall be so located with respect to the propellers that no portion of the pilot or controls lies in the region between the plane of rotation of any propeller and the surface generated by a line passing through the center of the propeller hub and making an angle of 5° forward or aft of the plane of rotation of the propeller.

(*C*) Closed cabins on aircraft carrying passengers shall be provided with exits as follows:

(1) Aircraft with a seating capacity of four persons or less shall be provided with an adequate and easily accessible door, but need have no other exit.

(2) Aircraft with a seating capacity for more than 4 persons, but not in excess of 15, shall be provided with at least one emergency exit or one suitable door in addition to the main door specified in item (1). This emergency exit, or second door, shall be on the opposite side of the cabin from the main door. If desired, an additional emergency exit may be provided in the top of the cabin, but such an installation will not obviate the necessity for an exit on each side.

(3) For aircraft with a seating capacity of more than 15 persons there shall be an additional emergency exit or door either in the top or sides of the cabin for every additional 7 persons or fraction thereof above 15, except that not more than 4 exits, including doors, will be required if the arrangement and dimensions are suitable for the purpose intended.

(4) Movable windows which provide an elliptical or rectangular opening 17 by 24 inches or a circular opening 24 inches in diameter, may be considered as emergency exits if their location and method of operation are approved.

(5) If the pilot is in a compartment separate from the cabin, passage through the cockpit shall not be considered as an emergency exit for the passengers.

(6) No passenger door shall be located in the plane of rotation of a propeller, nor within 5° thereof as measured from the propeller hub.

(*D*) Controls shall be so constructed or arranged as to prevent passengers or cargo from interfering with the course of flight of the airplane. A control column or stick located between a pilot and a passenger shall not be used unless the “throw-over” type of wheel control is incorporated.

(*E*) Baggage and mail compartments shall be designed for a definite maximum allowable weight of contents and shall be flight tested with this load. They shall bear a placard stating the maximum load for which the structural strength and flight characteristics are satisfactory. Suitable means shall be provided to prevent the contents of mail and baggage compartments from shifting.

(*F*) A metal identification plate shall be permanently affixed in a visible location in the pilot's cockpit or compartment of each airplane. This plate shall contain the manufacturer's name, the date of manufacture, the manufacturer's serial number, the model designation, and the type of engine.

(*G*) The manufacturer shall specify the fuel capacity of each fuel tank on the manufacturer's identification plate, or on, or adjacent to, the fuel shut-off valves in the pilot's cockpit.

(*H*) On all airplanes a placard shall be placed in full view of the pilot (or pilots) bearing the following words:

DO NOT EXCEED____ MILES PER HOUR TRUE INDICATED AIRSPEED AT ANY TIME

The limited gliding speed thus specified shall be 10 miles per hour less than the design gliding speed used in the stress analysis, as defined in section 14 (C). The actual air-speed-indicator reading to which the above placard speed corresponds shall be determined by calibration of the air-speed-indicator and noted on the placard.

(*I*) On all airplanes employing wing flaps, the placard described in paragraph (H) shall also bear the following words:

DO NOT EXCEED____ MILES PER HOUR TRUE INDICATED AIRSPEED WITH FLAPS EXTENDED

The speed thus specified shall be 10 miles per hour less than the design speed used in the stress analysis of the wing with flaps extended, as outlined in section 18 (D) (1) (b). The actual air-speed-indicator reading corresponding to the limited speed with flaps extended shall be determined by calibration of the air-speed-indicator and noted on the placard.

(*J*) Inspection openings of adequate size shall be provided for all vital parts of the aircraft.

Sec. 65. Engines

(*A*) Engines shall be of a type and design which is approved and assigned a power rating and speed rating by the Secretary of Commerce. *Engines for use in light aircraft as defined in section 2 (B) need not be approved but shall have ratings assigned by the Secretary.* The power and speed ratings assigned by the Secretary shall be used in all computations substantiating the airworthiness of the aircraft. Engines are not eligible for operation in excess of the rated speed except for the purpose of setting fixed or adjustable pitch propellers.

(*B*) Altitude engines are assigned a rated altitude in addition to the power and speed rating. At this altitude and above, such engines are suitable for continuous operation at full throttle. For operation below this altitude the pilot shall limit the output of the engines to the approved output for the flight condition involved. An absolute manifold-pressure gage or other suitable means shall be provided for this purpose.

(*C*) Requirements for the approval of engines are outlined in chapter I of Aeronautics Bulletin No. 7-G.

Sec. 66. Propellers

(*A*) Propellers shall be of a type and design which is approved and assigned a maximum power and maximum speed rating by the Secretary of Commerce. *Propellers for light aircraft as defined in section 2 (B) need not be approved.* In certain cases maximum engine bore limitations are also assigned to propellers. Propellers may be used on any engine provided that that approved power rating, speed rating, and bore of the engine are not in excess of the maximum rated power, maximum rated speed, and bore limitation of the propeller, and further provided that the vibration characteristics of the combination are satisfactory.

(*B*) Fixed or adjustable pitch propellers shall limit the engine speed at full throttle in level flight at the rated altitude of the engine to not more than 105 percent of the approved rated engine speed. The extra 5 percent permitted above the rated speed for these propellers is for the purpose of making available additional power during take-off and climb.

(*C*) Pending the development of suitable detailed requirements for controllable or automatic-pitch propellers, the controlling mechanism should be designed so that the pilot can prevent the engines from exceeding their rated speed at any time, or their approved output for the flight condition involved. The means provided for this purpose should be designed so as to minimize the attention required from the pilot in maintaining the proper pitch settings.

(*D*) Propellers shall have a minimum ground clearance of 9 inches when the airplane is in a horizontal position with the chassis deflected as it would be under the normal gross weight of the airplane. Propellers on seaplanes shall clear the water by at least 18 inches when the seaplane is at rest. A clearance of at least 1 inch shall be provided between the tips of propellers and the fuselage or any part of the structure. Surfaces near propeller tips shall be suitably stiffened against vibration.

(*E*) Requirements for the approval of propellers are outlined in chapter II of Aeronautics Bulletin No. 7-G.

Sec. 67. Fuel System

(*A*) Air-pressure fuel systems shall not be used. Only straight gravity feed or mechanical pumping of fuel is permitted. The system shall be so arranged that the entire fuel supply may be utilized in the steepest climb and at the best gliding angle and so that feed ports will not be uncovered during normal maneuvers involving moderate rolling or side slipping. If a mechanical pump is used, an emergency hand pump shall be installed. Hand pumps may also be used for pumping fuel from an auxiliary tank to a main fuel tank.

(*B*) No fuel tanks shall be placed closer to an engine than the remote side of a firewall. At least one-half inch clear air space shall be allowed between the tank and firewall. Surfaces of the tanks shall be ventilated so that fumes cannot accumulate in case of leakage. Each fuel tank shall be provided with a sump and drain located at the lowest point when the airplane is in a normal position on the ground. The main fuel supply shall not be drawn from the bottom of this sump. Each tank shall be suitably vented from the top portion of the air space. Such air vents shall be arranged so as to minimize the possibility of stoppage by ice formation. If two or more tanks have their outlets interconnected, the air space in the tanks shall also be interconnected to prevent differences in pressure at the air vents of each tank of sufficient magnitude to cause fuel flow between tanks. Where large fuel tanks are used, the size of the vent tubes should be proportioned so as to permit rapid changes in internal air pressure to occur and thereby prevent collapse of the tanks in a steep glide or dive.

(*C*) Fuel tanks shall be capable of withstanding an internal test pressure of 3 ½ pounds per square inch without failure or leakage. Fuel tanks of large capacity which have a maximum fuel depth greater than 2 feet shall be investigated for the pressure developed during the maximum applied acceleration with full tanks.

(*D*) A satisfactory gage shall be installed on all aircraft indicating to the pilot or flight mechanic the quantity of fuel in each tank while in flight. Where two or more tanks are interconnected and it is impossible to feed from each one separately, only one fuel-level gage need be installed. If a glass gage is used, it shall be suitably protected against breakage.

(*E*) All fuel piping and fittings shall be of sufficient size so that under the pressure of normal operation the flow is not less than double the normal flow required for full engine power. A test for compliance with this requirement may be necessary in certain cases. Bends of small radius and vertical humps in the line shall be avoided. Copper fuel lines which have been bent before installation shall be annealed. A flexible connection or its equivalent is required between the part of the fuel system attached to the engine and that attached to the primary structure of the airplane. Flexible hose connections shall have metal liners. Flexible fuel lines shall be suitably supported. The spacing of such supports depends on the type of fuel line used and, in general, varies between 6 and 12 inches.

(*F*) One or more strainers of adequate size and design, incorporating a suitable sediment trap and drain, shall be provided in the fuel line between the tank and the carburetor and shall be installed in an accessible position, so that the screen may be easily removed for cleaning.

(*G*) One or more positive and quick-acting valves that will shut off all fuel from the engine shall be within easy reach of the pilot or flight mechanic. In the case of aircraft employing more than one fuel supply, suitable provision shall be made for independent feeding from each supply. When fuel tanks are equipped with dump valves the operating mechanism for such valves shall be within convenient reach of the pilot or flight mechanic and the outlets shall be so located as to preclude the possibility of fire when they are operated.

Sec. 68. Lubrication System

(*A*) The oil capacity of the system shall be at least 1 gallon for every 16 gallons of fuel but shall not be less than the minimum specified for safe operation of the engine by the engine manufacturer. A suitable means shall be provided to determine the amount of oil in the system.

(*B*) Oil tanks shall be capable of withstanding an internal test pressure of 5 pounds per square inch without failure or leakage. They shall be suitably vented and shall be provided with expansion space which cannot be inadvertently filled with oil. This expansion space shall be at least 10 percent of the total tank volume, except that is shall in no case be less than one-half gallon.

(*C*) Oil piping shall have an inside diameter not less than the inside diameter of the engine inlet or outlet and shall have no splices between connections. The oil temperature should be measured at the engine inlet. Connections in the oil system shall be of a type satisfactory to the Secretary.

Sec. 69. Cooling System

(*A*) The cooling system shall be of sufficient capacity to maintain engine temperatures within safe operating limits under all conditions of flight during a period at least equal to that established by the fuel capacity of the aircraft, assuming normal engine speeds.

(*B*) Radiators shall be mounted so as to reduce vibration and eliminate strains causing distortion.

(*C*) Piping and connections shall conform to accepted standards and shall not transmit vibration to the radiator or the structure of the aircraft.

Sec. 70. Miscellaneous Power Plant Requirements

(*A*) The engine, ignition, fuel, oil, and cooling systems and all accessories normally included in an airplane power plant shall be so designed and installed as to reduce to a minimum the chances of failure to function while in the air, and insofar as is possible, to prevent fires during flight or in the event of a crash. Every effort should be made by the manufacturer toward fire prevention, in addition to compliance with the various specific requirements previously outlined. Engine installations and fuel and oil systems will be critically inspected for possible fire hazards and any questionable details or methods will be rejected as unsafe.

(*B*) Tanks, connections, and piping shall be made of materials, including their normal or inherent impurities, which will not react chemically with any fuel, oils, or liquids that are likely to be placed in the tanks.

(*C*) A firewall shall be provided unless the engine is mounted in a separate nacelle with no fuel tanks. Firewalls shall be made of a ferrous metal of suitable thickness, except that aluminum or aluminum alloy may be used for the outside sheets of an asbestos "sandwich." The firewall shall completely isolate the engine compartment and shall have all necessary openings fitted with close-fitting grommets or bushings. Adjacent inflammable structural members shall be protected by asbestos or some equivalent insulating material and provision made for preventing fuel and oil from permeating them.

(*D*) All cowling around the power plant and on the engine side of the firewall shall be made of metal and shall be so arranged that any accumulations of dirt, waste, or fuel may be observed without complete removal of the cowling. It shall fit tightly to the firewall, but openings may be provided if the airplane surface within 15 inches is protected with metal or other suitable fireproofing material. The cowling shall be completely drained in all attitudes of flight and on the ground, with separate drains provided for the parts of the fuel system liable to leakage. All such drains shall be so located as to prevent fuel or oil from dripping onto the exhaust manifold or any parts of the aircraft, or permeating any material of a cellular nature.

(*E*) Carburetor air intakes shall open completely outside the cowling, unless the emergence of back-fire flames is positively prevented, and should be suitably drained.

(*F*) Exhaust manifolds shall be constructed of suitable materials, shall provide for expansion, and shall be arranged and cooled so that local hot points do not form. Gases shall be discharged clear of the airplane structure and so that they will not blow back on the carburetor air intake, the pilot, or passengers, nor cause a glare ahead of the pilot at night.

(*G*) All filler openings shall be plainly marked with their capacity and the word "fuel", "oil", or the name of the cooling liquid, as the case may be.

(*H*) One or more accessible drains shall be provided at the lowest point on the fuel, lubricating, and cooling systems to drain completely all parts of each system when the airplane is in its normal position on level ground. Such drains shall discharge clear of all parts of the airplane and be equipped with suitable safety locks to prevent accidental openings.

(*I*) All engine controls shall be plainly marked to show their function and method of operation. Throttle control and ignition switches shall be easily accessible to the pilot and shall be so arranged as to afford a positive means of controlling all engines separately or simultaneously. A positive means for shutting off all ignition shall be readily accessible to the pilot. Throttle controls may be operated by an approved positive-action cable or wire control system.

Sec. 71. Electrical Equipment

(*A*) Electrical equipment shall be installed in accordance with accepted practice and suitably protected from fuel, oil, water, and other detrimental substances. Adequate clearance shall be provided between wiring carrying appreciable current and fuel and oil tanks, fuel and oil lines, carburetor, exhaust piping, and moving parts.

(*B*) Fuses shall be located so that they can readily be replaced in flight. They shall break the current in a generating system at a sufficiently small current flow to adequately protect the lights and other parts of the circuit.

(*C*) Batteries shall be easily accessible and adequately isolated from fuel and oil systems. Adjacent parts of the aircraft structure shall be protected with a suitable acid-proof paint if the battery contains acid or other corrosive substance. All batteries shall be installed so that spilled liquid will be suitably drained or absorbed without coming in contact with the airplane structure. If the battery is completely enclosed, suitable ventilation shall be provided.

Sec. 72. Equipment and Instruments

The equipment and instruments required, which shall be serviceable and in operating condition, are:

(*A*) *Equipment.—*

(1) Fire-extinguishing equipment, of design approved by the Secretary. Fire-extinguishing equipment approved by the Underwriters Laboratories is satisfactory. Cabin airplanes carrying passengers shall be equipped with at least one portable extinguisher having a minimum capacity of 1 quart, accessible to the passengers. One pint extinguishers may be used in airplanes employing 40 horsepower or less.

(2) Safety belts or their equivalent for pilot and passengers, which shall be of an approved design as outlined in Aeronautics Bulletin No. 7-F.

(3) Electrical equipment and flares required in Aeronautics Bulletin No. 7-F, for the type of flying for which the airplane is to be used.

(4) Aircraft and engine log books.

(5) Rigging information for airplanes with wire-braced wings, either in the form of a sketch or listed data, which shall include sufficient information to permit proper rigging.

(*B*) *Instruments.—*

(1) Tachometer for each engine.

(2) Oil-pressure gauge where oil-pressure systems are employed.

(3) Water thermometer for water-cooled engines and oil thermometer for air-cooled engines.

(4) Altimeter.

(5) A manifold-pressure gauge or equivalent for each altitude engine.

(*C*) *Compass requirements.—* Any airplane flying across country 100 or more miles or operating over large bodies of water beyond the sight of land, shall be equipped with a compass.

Sec. 73. Performance

(*A*) The following characteristics are required with full load in still air for all land planes and amphibians. For seaplanes and flying boats the values in parentheses may be used when specified.

(1) Land at a speed not exceeding 65 miles per hour, except that airplanes which are neither designed nor used to carry passengers shall land at a speed not exceeding 70 miles per hour.

(2) Take-off within 1,000 feet at sea level in calm air. (Seaplanes and flying boats shall take-off without excessive run under normal conditions.)

(3) The climb in feet the first minute after leaving the ground or water shall be at least 8 (6) times the theoretical stalling speed in miles per hour, but not less than 300 (250) feet.

(4) Multiengined aircraft shall be capable of maintaining level rectilinear flight with full load with any one engine out of commission and the remaining engines not exceeding their rated power or rated speed at an altitude of not less than 2,000 (1,000) feet above sea level, except that airplanes which are equipped with suitable valves for the rapid discharge of fuel shall be capable of maintaining level rectilinear flight at 2,000 (1,000) feet with full load minus the amount of fuel which can readily be dumped through such valves. In the latter cases at least 10 effective gallons of gasoline per hundred horsepower of the remaining engines shall be retained in the aircraft and this fuel shall be immediately available for use in the engines which are still functioning. In the case of airplanes with variable pay load, dependent upon the amount of fuel carried, compliance with this regulation shall be demonstrated with the maximum permissible pay load on board.

(5) Multi-engined aircraft shall be able to climb from sea level to an altitude of 1,000 feet with one engine dead, all engines being used to attain the minimum take-off speed.

(*B*) *Transport aircraft.*— See Aeronautics Bulletin No. 7-J for special requirements for transport aircraft.

Sec. 74. Flight Tests

(*A*) The flight tests, in addition to demonstrating the performance required by section 73, shall be such as to demonstrate the balance, stability, maneuverability, and general flying and taxiing characteristics of the airplane. Prior to, or at the time of, presentation of an airplane to the Department for flight tests the manufacturer shall submit to the inspector making the tests a detailed report of the manufacturer's flight tests on the airplane involved. The report submitted shall be signed by the manufacturer's test pilot and shall indicate that the aircraft has been fully test flown, including all maneuvers required herein.

(*B*) Under all loading conditions all airplanes shall have adequate control in a landing at minimum landing speed with tail down and power off.

(*C*) Under all load and power conditions, all airplanes shall be longitudinally, laterally, and directionally stable.

(*D*) At engine speeds between the minimum required for level flight and 90 percent of the rated engine speed all airplanes shall be longitudinally balanced and stable in level flight. All airplanes shall preferably be longitudinally balanced and stable in level flight at full power but will be acceptable if, with the stabilizer set for balance in level flight at 90 percent rated engine speed, they assume and maintain a stable climb at engine speeds above 90 percent without stalling or falling out of control. With power off, all airplanes shall be longitudinally balanced and stable at some speed between 120 and 140 percent of the stalling speed at sea level. All the above conditions shall apply to all load conditions. An airplane will be considered unstable if, in longitudinal stability tests, the amplitude of the oscillations increases. The maximum limited speed for which it is placarded shall not be exceeded during any oscillation.

(*E*) All airplanes shall be able to recover from a 6-turn spin in no more than 1 ½ additional turns with controls neutral, power off, and with the stabilizer set for balance at cruising speed in level flight with the particular load. During the spin the control surfaces shall exert no back pressure on the control column. When a spin is obtained by movements of the controls which the inspector considers to be abnormal and unjustifiably severe for operating maneuvers, the use of elevator and rudder will be permitted for recovery after six turns. An airplane of this type will be acceptable insofar as spin characteristics are concerned provided recovery can be made in not more than 2 additional turns with ailerons in neutral, and further provided that the normally obtained spin either conforms with the normal requirements or is impossible to perform. An airplane of more than 4,000 pounds gross weight will not be subjected to the spin test unless the flight characteristics of the particular airplane are, in the opinion of the Secretary, such as to indicate the advisability of a spin test.

(*F*) *The maximum limited speed for which the airplane is placarded shall not be exceeded in any flight test.*

(*G*) The seaworthiness of flying boats and large seaplanes shall be demonstrated by appropriate rough-water tests.

(*H*) No surface shall show any signs of flutter or appreciable vibration in any normal attitude or condition of flight at any speed between the minimum flying speed and the maximum limited speed.

(*I*) The applicant shall provide a person holding a transport pilot license to make the flight tests, but the Department of Commerce representative may pilot the airplane during such parts of the tests as may be deemed advisable.

(*J*) No persons other than the crew necessary to conduct the flight shall be carried during the flight tests. Sand or similar ballast shall be carried as pay load.

(*K*) Ballast may be used to enable airplanes to comply with the flight test requirements as to longitudinal stability, balance, and landing, with the following provisions:

(1) Ballast shall not be used for this purpose in aircraft having a gross weight of less than 10,000 pounds, except that airplanes having a seating capacity for more than 8 passengers may use an amount of ballast not exceeding 20 pounds per passenger carried in the specific loading condition in question.

(2) It shall be demonstrated that the specific airplane is capable of making a safe landing without ballast, passengers, or cargo.

(3) In airplanes for which ballast is utilized a placard shall be placed within full view of the pilot, stating the minimum weight of ballast which should be carried when the airplane is flown without pay load and the place where the ballast must be located.

(4) The place for carrying ballast shall be properly provided and plainly marked.

(5) If the space provided for carrying ballast is some place other than one in which normally an equal amount of pay load would be carried, due allowance for the ballast weight shall be made in the design of the airplane structure.

(*L*) During the flight tests, the limiting permissible fore and aft positions of the center of gravity within the range made possible by the arrangement of loads, shall be determined. The vertical movement of the center of gravity may be disregarded except in specific cases where unusual circumstances indicate the advisability of determining its effect on the flight characteristics of the airplane. These positions shall be located with respect to the mean aerodynamic chord by actual weighing of the airplane in the presence of the Department representative making the test, and by suitable computations. *A complete report of the weighing and the necessary computations shall be prepared and submitted by the manufacturer and the report shall be signed by the Department representative.* Later requests for approval of changes in internal equipment or arrangement, when supported by certified weight statements and computations, may be checked against the original balance results to determine whether or not additional flight tests will be necessary.

Sec. 75. Weights

(*A*) In general, the design gross weight of an airplane will equal the sum of the weights of the empty airplane, total fuel capacity, total oil capacity, crew, and total pay load.

(*B*) The weight empty includes the weights of all items except fuel, oil, crew, and pay load. Pay load consists of passengers, baggage, and cargo. Baggage includes hand luggage and other loose equipment, such as parachutes, tools, and anchors. Useful load is the difference between allowable gross weight and weight empty. Weights are calculated on the following bases: Gasoline, 6 pounds per gallon; lubricating oil, 7.5 pounds per gallon; crew and passengers, 170 pounds per person; parachutes, 20 pounds each.

(*C*) In the case of airplanes for which the allowable gross weight is not sufficient to permit carrying simultaneously the full fuel and full pay load, approval will be granted under the following conditions:

(1) The available gasoline capacity with full pay load shall be at least 0.15 gallon per rated engine horsepower. The oil capacity shall be sufficient for this amount of fuel. If the fuel used is other than gasoline, the available fuel capacity with full pay load shall be sufficient for a 2-hour flight at cruising speed.

Sec. 76. Location and Dimensions of Identification Marks

(*A*) Identification marks shall be affixed to all aircraft in accordance with Aeronautics Bulletin No. 7, section 41.

Sec. 77. Transport Aircraft

Special requirements for transport aircraft are outlined in Aeronautics Bulletin No. 7-J.

Figure 1.— Power loading factor.

Figure 2.— Span distribution (no tip loss).

Figure 3.— Span distribution (with tip loss).

Figure 4. — Tail load chord distribution — forward portion.

Figure 5. — Tail load chord distribution — rear portion.

Figure 6.— Aileron chord distribution.